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L. M. Vorob'ev

SPACECRAFT NAVIGATION

TRANSLATED FROM RUSSIAN

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L.M. VOROB'EV

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The tremendous successes in rocketry which made possible flights into space have greatly enhanced the Soviet public's interest in the universe.

In this book a series of questions are considered: what is the universe; what will man encounter when he penetrates it; what velocities would be required; what should be the trajectories and the most advantageous times for flights to the moon, to other planets of the solar system, and to the nearest stars; what means can be used to determine the position of a spacecraft traveling in space.

The book is well illustrated. It reflects the latest Soviet and foreign achievements in the field of astronautics and astronomy.

The book is intended for a wide circle of readers interested in questions related to the conquest of space.

PREFACE

The beginning of the second half of the twentieth century has been marked by great developments in science and technology. The penetration into the depth of the nucleus, which was crowned by the realization of controlled nuclear reactions, on the one hand, and the exploration of space on the other, are perhaps mankind's most important achievements in this period. Great interest in many problems related to the universe has been aroused in a wide circle of the Soviet public, and especially among the youth. Indeed, can one remain indifferent to the stories of heroic countrymen, hero-cosmonauts, who have made these flights into space? Is it possible to read without excitement even the miserly brief lines of newspaper announcements of flights by artificial satellites and space rockets, of the unprecedented scientific experiments in space and of the outstanding scientific discoveries that have been made?

Space travel has always been a subject of special popular interest. A proof of this is the publication of so many works of science fiction. Real accomplishments have now come to replace fantasy. Man has penetrated into the near cosmos, and automatic interplanetary stations with numerous instruments have been launched from the earth to the moon and to the nearest planets — Venus and Mars.

In his eternal aspiration to know the secrets of the universe, man is not satisfied with the results so far achieved. Piloted spacecraft will fly to the moon and to neighboring planets, and, perhaps within the lifetime of the present generation, spaceships from the earth will be sent to the remote planets of the solar system and even beyond its boundaries.

However, the flight of man even to the nearest celestial body — the moon — requires the solution of very complicated scientific and technological problems and careful preparation.

In a talk with the delegates of the Third World Convention of Journalists on 25 October, 1963, N.S.Khrushchev answered a question on the possibility of a flight by Soviet cosmonauts to the moon as follows:

"At present we are not planning flights of cosmonauts to the moon. Soviet scientists are working on this question, studying it as a scientific problem, and conducting the necessary investigations. I have read the report that the Americans want to land a man on the moon in 1970... We do not wish to compete in the sending of men to the moon without careful preparation; it is clear that no good would come out of a competition of that kind, which, on the contrary, would be harmful, since it could result in the loss of human lives... Before man can make a successful flight to the moon much work and sound preparation will be required."

In connection with the huge successes in astronautics recently achieved in the USSR and abroad, a large number of special articles and popular essays have appeared which analyze the results obtained and consider the immediate and long-term prospects for man's penetration into the depths latest

Methods of solving the most important problems of cosmic navigation are described in a number of recently published works. Among them one should particularly note the works of Aleksandrov and Fedorov /2/*, Seleznev /29/, Levantovskii /19/, and Erike /41/, to which the author refers readers who are interested in problems of space navigation and who wish to study this new highly-interesting field of science in greater detail.

This book is designed for those first being introduced into this subject. In it the author has endeavoured to consider the various aspects of navigation, and to show how the fundamental questions involved in the navigation of space vehicles have been solved. However, it is not a study of space navigation problems and not a textbook; the author has therefore tried to present his material as simply as possible. It remains for the reader to judge how far he has succeeded in these aims.

The book is intended for a wide circle of readers interested in the latest achievements in the conquest of space, and particularly for young people eager for knowledge, and if the reader finds in this work useful information, if his reading arouses in him a lively interest in the questions discussed, the author will consider his task fulfilled.

The author takes this opportunity to express his profound thanks to Engineer-Colonel V. P. Seleznev, Doctor of Technical Sciences, who made a number of valuable comments and suggestions which were extremely helpful in the preparation of the manuscript.

* Numbers in oblique brackets relate to the bibliographical references at the end of the book.

INTRODUCTION

On 4 October, 1957, in the USSR, for the first time in man's history an artificial earth satellite was successfully launched into orbit. Man had created an artifical moon; a velocity had been attained which made it possible to realize prolonged flights in space.

We have already become accustomed to the vigorous tempo of modern scientific and technological developments. We perceived the progress of aviation and the rise in the velocity of airplanes as usual phenomena, but this event profoundly excited us. Indeed, for the first time in the history of mankind, a velocity had been attained (over 28,000 km/hr!) which exceeds the velocity of modern jet planes by a factor of more than 10. A terrestrial body created by man's hands reached a height of about 1000 km. The dream of reaching space had been realized by the Soviet people.

The 20th century is the century of electricity, atomic energy, cybernetics, computers, automation, and new plastic materials. Obviously, it is not less justified to call it the century of the conquest of space, the century of man's escape beyond the boundaries of the "air ocean" — the atmosphere of the earth. This is confirmed by the remarkable achievements in the conquest of space and the unparalleled tempo of the development of astronautics — the science and practice of flights in space.

On 3 November, 1957, the second artificial earth satellite was launched into orbit by the USSR. The USA, the foreign country leading in scientific and technological developments, put its first artificial earth satellite — "Explorer I" — into orbit only on 31 January, 1958.

With the launching of the artificial earth satellites there arose the possibility of direct experimental investigation of the upper layers of the atmosphere and of outer space. Astronomy, geophysics, meteorology, and other sciences received a new, extremely powerful instrument of research. Even the first launchings led to outstanding scientific discoveries. The radiation belts of the earth were discovered by direct experiment, the values of the parameters of the upper layers of the earth's atmosphere were determined and extensive investigations on the earth's magnetic field were carried out. Soviet, and later also American cosmonauts demonstrated the possibility of prolonged human flight in outer space under conditions of weightlessness.

The launchings in 1957-1958 of the Soviet earth satellites (sputniks) made it possible to accumulate the necessary material for flights deeper into space. On 2 January, 1959, a Soviet rocket was launched in the direction of the moon. In this case the second escape velocity — the minimum velocity necessary to get free of the gravitational field of the earth — was attained. On 12 September, 1959, a second rocket to the moon was successfully launched, and on 14 September at 0 hr 02 min 24 sec Moscow time it reached the surface of the moon. For the first time in history a flight had been made from the earth to another celestial body. On 4 October, 1959, the third Soviet cosmic rocket was launched to the moon, and unique photographs of the part of the lunar surface which is not visible from the earth were obtained. These made it possible to prepare an atlas of the moon and a lunar globe.

While 1959 can be called the year of Soviet lunar flights, 1961 can be called the year of the first manned space flights and the first interplanetary flights. The dates 12 April and 6 August, and the names Yu. A. Gagarin and G. S. Titov will forever be remembered in the history of mankind. People will never forget the feat of Soviet cosmonauts, the first to pave the way to the stars. On 12 February, 1961, for the first time in history, a Soviet automatic interplanetary station was launched to Venus.

On 20 February and 24 May, 1962, the American astronauts J. Glenn and S. Carpenter made their orbital flights. This was followed by a new achievement of the Soviet people, an oustanding success of Soviet science and technology — the orbital flight of several days of the two cosmonauts A. G. Nikolaev and P. R. Popovich in August 1962; then the 1 November launching of the automatic interplanetary station "Mars I" — the next attempt to penetrate into the depths of the solar system. The "multi-day" double flight in the summer of 1963 in the manned satellites "Vostok V" and "Vostok VI" of V. F. Bykovskii and of the first woman cosmonaut V. V. Tereshkova — a new example of heroism and courage — was a further outstanding success of Soviet science and technology. The Soviet launching of the maneuverable spacecraft "Polet I" on 1 November, 1963, opened a new page in the conquest of space.

Man's indomitable will, supported by the greatest achievements of his mind, will every day penetrate further and further beyond the limits of the earth's atmosphere.

The science dealing with the study of the problems of interplanetary space flights is called astronautics. It includes a whole series of independent scientific disciplines. A constituent part of astronautics is the science which can be called space navigation. This science has not yet been definitively formulated. The scope of problems which have to be studied has not yet been defined, its purpose and problems are not yet clearly delineated. However, this science is an essential and important one, particularly with the beginning of flights into the depths of the universe.

It would seem that the fundamental problems to be considered by this science will be those connected with the choice of the most advantageous flight trajectories of spaceships, as well as the problems connected with the guidance of spaceships along their prescribed trajectories. The solution of these questions requires first of all a study of the universe. Astronauts, spacecraft captains, and pilots of future spaceships will need to know the navigational structure of that part of the universe in which they have to fly their interplanetary ships.

Like seamen and land travelers studying geography, astronauts should have a working knowledge of cosmology — one of the branches of astronomy which deals with the structural regularities of the universe — and some other other branches of astronomy. In fact, there will be take-off and arrival points for spaceships not only on the earth, but also on other planets and celestial bodies. The trajectories of these ships will pass through gravitational fields of other celestial bodies and therefore the form and parameters of the trajectories of the spaceships will be determined by the physical characteristics of those celestial bodies, primarily their masses. The choice of the safest path for a spaceship also requires the study of the position of meteor streams and dangerous zones of intense cosmic radiation.

The solution of a very important problem of navigation — determination of the position of the spacecraft in space — would be impossible without knowledge of such subjects as the dimensions of celestial bodies, their apparent brightness, their relative position and so on.

Finally, the cosmic traveler will need to know what to expect on arrival at the celestial body to which he is heading; what his weight will be there, i.e., what the gravitational acceleration on the surface of the celestial body is; what kind of atmosphere the celestial body has; the condition and temperature of its surface.

This is far from being a complete list of the most important problems in the field of cosmology and other branches of astronomy, which astronauts must get acquainted with when planning to fly to other planets, to other worlds. Accordingly, the first chapter of this book gives a short description of space and of the nearest celestial bodies; the second chapter deals with the question of orbits and trajectories for reaching the nearest celestial bodies and gives a comparative evaluation of them; and in the third chapter various possible methods and devices for resolving the principle problems as well as some of the subsidiary questions involved in the navigation of interplanetary ships and low-orbit spacecraft are described.

Chapter I

THE LIMITLESS EXPANSE OF THE UNIVERSE

§1. Constellations and Stars

On a clear night we can see a majestic picture of the starry firmament. It is so familiar that we no longer admire its beauties or even pay attention to it at all. However, one need only look once more at the stars and constellations and compare the colors of some of the brightest stars, to realize their unique beauty. On a particularly dark night, far from the city or large populated areas an even more majestic sight lies before one's eyes. The large stars then seem quite close and beyond them a multitude of tiny stars form what appears to be a latticework screen.

This view approximates that seen from a spaceship, except for one feature of the astronaut's view that is unavailable to a terrestrial observer: the dazzling white-hot sun seen against a background of black sky and bright stars.

From the most ancient times the stars have served as a reliable means of guiding navigators; it was by the stars that the course of the ship and its position were determined. Even today, observations of the stars are widely used in marine and aerial navigation. It can be assumed that in navigation of spaceships, methods based on the measurement of the positions of the stars will also find a wide application.

Observation of the starry firmament shows us that the brightest stars form groups of characteristic, easily remembered shapes. For example, in midwinter, soon after nightfall we can see near the southern part of the horizon a group of bright stars forming a trapezoid in the center of which is seen a line of three stars of almost equal brightness.

These star groups were called constellations by the ancient civilized peoples. Ptolemy (2nd century B. C.) already mentioned 48 constellations. The constellations received proper names, taken from legends and myths, in particular from the Greek mythology. The group of stars mentioned above belongs to the constellation Orion. Everyone is familiar with the constellation of the Great Bear [Ursa Major]. There are the constellations Auriga, Gemini, Leo, Scorpio, Aquila, Andromeda, Vela, and so on.

In astronomy the same system of division of the celestial sphere into constellations is used now as was used by the ancient Greeks. The important difference is that now constellations are understood to be not mere groups of stars, but sections of the starry firmament. At the present time the whole sky is divided into 88 such sections — the constellations.

The stars in the constellations are denoted by letters of the Greek alphabet in the order of decreasing brightness (α, β, γ, δ , and so on). In addition, the brightest stars have proper names. Thus, the star α Andromedae (i. e., the brightest star in the constellation Andromeda) is called

Alfaretz, the star α Ursa Minoris — the North Star or Polaris, the star α Geminorum (i. e., the brightest star in Gemini) — Castor, and β Geminorum — Pollux. The brightest star in the sky — α Canis Major — is called Sirius.

Because of the great distances to the stars and the small apparent angular velocity of their motion relative to one another, even astronauts on flights to remote planets of the solar system will observe the usual terrestrial picture of the stars and their constellations. Moreover, from the spaceship it will be possible to see them during the whole flight, since there will be no cloudiness and the dazzlingly bright sun and moon would not obstruct observation of the stars.

The use of astronomical methods of spacecraft navigation requires the selection of stars for the solution of a given navigation problem. The positions of the stars, their spectral characteristics and their brightness are the basic factors to be considered in selecting the stars to be used for space navigation.

The apparent brightness of the star is denoted by its stellar magnitude. Even in ancient times attempts were made to classify stars by their apparent brightness. Two thousand years ago, Hipparchus proposed to divide the stars according to their brightness into stellar magnitudes.* This method of expressing the brightness of stars by the stellar magnitude is still used today.

The classification of stars into stellar magnitudes, due to certain features of our eyesight, is based on the Weber-Fechner law. This law is common for all the sensory organs of man and is used by physicists to estimate the intensity of sounds and noises. For the eyesight this law can be formulated as follows: *if the strength of the light source varies in a geometrical progression, then the corresponding sensation of brightness varies in an arithmetic progression.*

The scale of apparent brightness was defined so that the ratio of the brightness of a star of a given magnitude (E_m) to the brightness of a star of the next lower magnitude (E_{m+1}) is constant:

$$\frac{E_m}{E_{m+1}} = \text{const.}$$

Let us denote this ratio by n. Then, stars of the second magnitude are weaker than stars of the first magnitude by a factor of n. Stars of the third magnitude are weaker than stars of the second magnitude, also by a factor of n, and so on. Compared with the brightness of a star of the first magnitude, the brightness of a star of the second magnitude is lower by a factor of n^1, the brightness of a star of the third magnitude is lower by a factor of n^2, of the fourth magnitude by a factor of n^3, of the fifth magnitude by a factor of n^4, and so on.

Study of ancient star catalogues has shown that for all the observers the ratio of the brightness of stars of two adjacent stellar magnitudes was maintained quite accurately as approximately 2.5. It was determined that on the average $n = 2.512$. Thus, a star of any magnitude shines with a luminosity approximately two-fifths that of the stars of the preceding stellar magnitude. For a more accurate notation of the apparent brightness of stars, the stellar magnitudes are expressed not only by integers, but also by decimal fractions, and the brightest celestial bodies even have a negative stellar magnitude.

* [Ptolemy is usually credited with inventing this method of classifying the stars.]

A man with an average eyesight can see stars up to the sixth stellar magnitude inclusive. It is calculated that there are about 4800 such stars in the sky /17/. According to some other data, the number of stars visible by the naked eye is estimated to be 5720 /18/. By means of modern optical instruments it is possible to observe stars up to the 21st or 22nd stellar magnitude. The number of stars up to the 21st stellar magnitude in the sky is estimated to be approximately 889 million.

The apparent brightness of artificial space objects is also estimated by stellar magnitude. Thus, in the announcement on the automatic interplanetary station "Mars I" of 5 November, 1962, it was stated that, according to photographs, the station and its carrier rocket were seen on the background of the night sky as stars of the 14th and 13th stellar magnitudes, respectively.

It is quite difficult to determine the zero stellar magnitude in the scale of star magnitudes. However, if it is agreed that a certain star has a definite stellar magnitude, then the magnitudes of other celestial bodies can be determined with respect to this reference star.

In astronomic practice, the stellar magnitude of celestial bodies is determined by means of special instruments called photometers. The luminosity of a celestial body is compared with the luminosity of a star whose stellar magnitude is determined or known. Sometimes an artificial star is used as a standard in the instrument for this purpose.

The stellar magnitude of an international candle, placed at a distance of 1 km, is 0.8. The star magnitudes of some celestial bodies are as follows: The brightest star, Sirius, has a negative stellar magnitude of minus 1.6. The stellar magnitude of the brightest stars of the northern sky are: Vega, 0.1; Fomalhaut, 1.3; Polaris, 2.1. It is interesting to note that the stellar magnitude of the sun is minus 26.8. the moon in the first and last quarters is minus 9, and the full moon is minus 12.6. The sun is brighter than the brightest star Sirius by approximately a factor of 10 billion.*

Travelers to other planets will observe variation of the apparent brightness of planets. There will be an increase in the brightness when approaching the planet and a decrease going away. Future travelers beyond the limits of the solar system will also be able to observe a variation of the brightness of the sun and of the nearest stars. Thus, from the boundaries of the solar system the sun will be seen as a star of minus 4 stellar magnitude. This is equal to the maximum brightness of Venus as observed from the earth.

Look again carefully at the starry sky and compare the colors of the bright stars. The red stars such as Arcturus, Aldebaran, and Antares differ from the bluish white stars like Rigel, Deneb, and Vega. The colors of stars vary (as the colors of solid bodies heated to different temperatures) from cherry red to white and even a bluish color. Star colors are classified by numbers according to a special scale. Thus, the extreme white blue and dark red colors are estimated respectively by minus 2 and 10. Intermediate colors are estimated by intermediate numbers. Thus, white corresponds to 0, and orange to 7.

In space flight, cosmonauts may observe an interesting picture of variation of the colors of celestial bodies (the Doppler effect**). When

* [The magnitude of the sun is −26.8. Sirius is −1.6. The difference is 25.2 magnitudes or a factor of $2.512^{25.2}$.]

** Doppler, C.J. (1803–1853) – An Austrian scientist. The effect called by his name was discovered in 1842.

flying at a high velocity towards the celestial body, it will appear bluer, and when flying away, it will appear redder.

In the navigational systems of spacecraft, automatic direction finding of stars will probably be effected by means of photoelectric devices. The apparent luminosity of stars as seen by such means is also measured in stellar magnitudes, and is termed the photoelectric stellar magnitude. Compared to the human eye, photoelectric devices have a different sensitivity to rays of different colors, and therefore the photoelectric stellar magnitude of a star differs from its visual stellar magnitude.

§2. The Metagalaxy and the Galaxy. The Sun and the Solar System

Space navigation should be based on the data of cosmology — a science dealing with the study of the structure of the universe. Astronauts and pilots of spaceships should have a sound knowledge of the structure of that part of the universe through which they have to navigate their spacecraft. Let us follow the astronauts and get acquainted, though in general lines, with the basic laws of the structure of the universe.

The study of the universe is mainly made by means of optical instruments such as telescopes. Very recently, radiotelescopes of various types have come into use, which make it possible to receive extremely weak signals of cosmic sources of radio emission. Finally, in the last few years, it has become possible to use satellites equipped as automatic interplanetary stations for this purpose.

The primarly fact impressed upon us by the study of the universe is the tremendous distances between the celestial bodies. For this reason, in astronomy, we do not use the ordinary units of length as the meter and kilometer but larger units such as the astronomical unit, light year, and parsec.

The astronomical unit (a.u.) is equal to the average distance from the earth to the sun. Until recently the astronomical unit was considered to be 149,500,000 km. This unit of length serves mainly for measurements within the solar system. Thus, the average distance from the sun to Mercury is 0.387 a.u., from the sun to Jupiter, 5.2 a.u., and so on.

The problems of space navigation, and the choice and calculation of trajectories of interplanetary ships, require a very accurate knowledge of the value of the astronomical unit. Failing this, the deviation of the trajectory of a spaceship from its destination could reach many tens of thousands of kilometers, and the flight task would not be accomplished. The problem was that measurements of the astronomical unit by different methods gave results differing from one another by hundreds of thousands of kilometers.

In 1961 and 1962, for the purpose of improving the accuracy of the astronomical unit as well as solving some other scientific problems, Soviet scientists succeeded in probing Venus with a radar beam. This new achievement made it possible to determine more accurately the value of the astronomical unit. According to the latest measurements, the astronomical unit is equal to 149,599,300 km with a possible error of ± 2000 km. This amounts to approximately 1 part in 75,000 of the measured distance. This outstanding result may be considered as an important contribution by Soviet science towards the future conquest of interplanetary space.

The astronomical unit is a huge unit of distance, but the other units, the light year and parsec, are much larger.

The light year is the path traversed by a light ray in interplanetary space during a year. This measure of length is expressed in kilometers by the number 9,460,000,000,000 ($9.46 \cdot 10^{12}$), i.e., a light year is equal to about 9.5 trillion km.

But even the light year is a relatively small unit of distance in astronomy. The distances to stellar systems are expressed by a number of light years followed by many zeros. Even the nearest stellar system, the nebula of the constellation Andromeda, is approximately a million light years away, and therefore the distances to stars and stellar systems are expressed in terms of larger units — parsecs.

The parsec (ps) is the distance at which the radius of the earth's orbit is subtended by an angle of 1 second. The angle subtended at a star by the radius of the earth's orbit is called the annual parallax of the star. The word parsec is formed by a combination of two words — parallax and second. The parsec is equal to 3.26 light years, or $30.8 \cdot 10^{12}$ km.

The distance to the nearest star, Proxima in the constellation Centaurus, is 1.31 ps or 4.3 light years. The distance to Sirius is 2.67 ps or 8.7 light years.

However, even the parsec is not the largest unit of length. To measure distances to the remotest stars and star systems, and to measure their dimensions, even larger units of length are used. These are the kiloparsec (kps) and megaparsec (mps), which equal one thousand and one million parsecs, respectively.

Astronomy is an ancient science, and the huge mass of data compiled makes it possible to draw accurate conclusions on the structure of that part of the infinite universe accessible to our survey. By means of modern powerful telescopes it is possible to study stellar systems which are two to three billion light years away. Even larger possibilities are opened by the methods of radio astronomy. By means of modern radio telescopes, we have succeeded in "penetrating" into the universe to distances of up to ten billion light years. Considerably more modest, for the time being, are the results of direct sounding of space by means of automatic interplanetary stations. Measurements have been made several tens of millions of kilometers from the earth — between the orbits of Venus and Mars — but even these first begininngs made it possible to unveil many secrets of the universe.

The part of the universe accessible to observation constitutes an agglomeration of stellar systems or galaxies (Figure 1), called the metagalaxy. At the present time, over 100 million galaxies have been discovered. Indications have been observed that, at the limits accessible to modern observation, there is a concentration of galaxies. This is, apparently, the center of the metagalaxy.

In one of the stellar systems, called "the Galaxy"* our solar system is situated. Our galaxy is lens-shaped. It is thickest at the middle and thins out toward the edges (Figure 2). Its diameter is about 85,000 light years ($800 \cdot 10^{15}$ km).

* From the Greek word "galaktikos," which means "milky." Hence the Milky Way as a descriptive name of our Galaxy.

FIGURE 1. Photograph of the galaxy in the constellation Andromeda, obtained by Miller (USA)

The reddish-yellow color of the central part appreciably differs from the bluish color of the spiral arms.

Our galaxy is not uniform. It consists of individual stars of different types, stellar clouds, star clusters, stellar associations, gaseous and dust nebulae, clouds of interstellar gas, diffuse cosmic dust, and individual atoms of chemical elements.

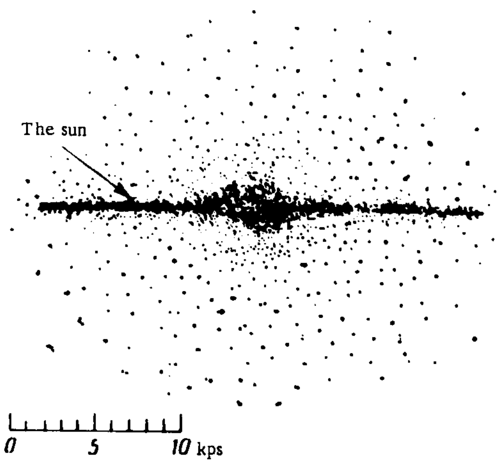

FIGURE 2. Lateral view of the Milky Way galaxy as it would appear to an observer situated in the plane of the galaxy (scale in kiloparsecs)

Star clouds are huge spaces filled with individual stars. In these [unlike star clusters] the stars are distributed at random. Star clusters — these star groups are found in the constellations of our galaxy (the Milky Way) and are of two kinds — open or galactic, and globular (Figures 3 and 4). The first are distinguished by sparse concentration of stars in the center of the cluster, the second by a dense concentration of stars there. In the galactic clusters, the number of stars is estimated from hundreds to several thousands. Another example of a galactic cluster is Pleiades. To the unaided eye from six to eleven stars are visible, while with a telescope it is possible to see hundreds of stars. One of the globular clusters, the cluster in the constellation Hercules, is seen as a nebulous star of approximately the sixth stellar magnitude. Only by means of a powerful telescope can we see it as a whole cluster of stars in the form of a sphere, strongly concentrated towards its center. In this star cluster there are hundreds of thousands of stars, of which only the brightest are seen. Stars of lesser luminosity, even those as bright as the sun, are not seen.

Star associations, discovered in 1947 by Academician V. A. Ambartsumyan, have a common origin; the process of formation of these stars occurred relatively recently and very likely is taking place even now. In associations, the stars are not as crowded as in star clusters. Clusters are unstable as the mutual attraction between them is negligible, and the tremendous attraction of the galaxy as a whole tends to scatter any cluster of stars.

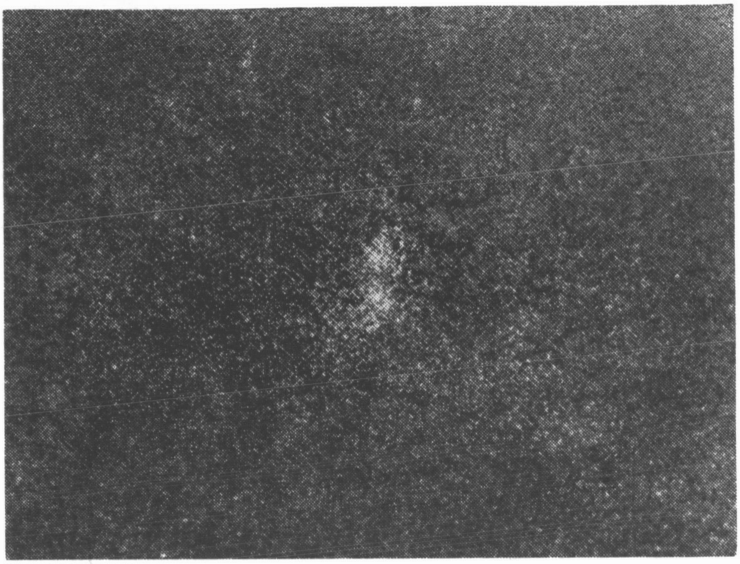

FIGURE 3. The galactic star cluster in the constellation Perseus

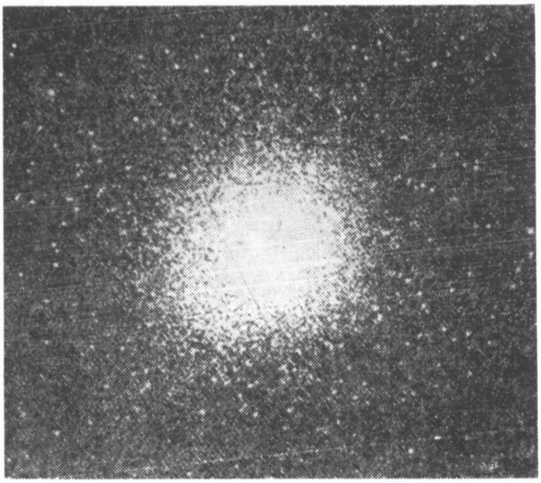

FIGURE 4. The globular star cluster in the constellation Centaurus

The existence of clouds of cosmic dust in interstellar space not only weakens the apparent luminosity of stars and causes them to appear red but also hides huge regions of our galaxy from us. Dark clouds or dark nebulae on the star sky are clouds of cosmic dust (about 10,000 tons of it falls daily on our planet) and gas, which completely hide the light of remote stars. Thus, the central parts of our galaxy are hidden from us by dark nebulae.

In addition to dust, there exists in interstellar space a highly rarefied gaseous "atmosphere" consisting mainly of atoms of hydrogen, as well as a certain quantity of atoms of helium, oxygen, nitrogen, sodium, calcium, and certain molecules (CH, NH, H_2O). There are also clouds of individual gases. The average gas density in our galaxy is negligible, amounting to only several atoms per cubic meter of space. However, the total mass of the gas is large, being almost equal to the mass of all the stars of the galaxy.

The galaxy rotates about an axis perpendicular to its plane, but not as a solid body. The motion of the stars in the galaxy resembles the motion of the planets about the sun; the farther from the center of rotation, the slower the motion.

The sun is situated almost exactly in the plane of the galaxy at a distance of about 23,500 light years from its center (see Figure 2). It moves in its orbit about the center of the galaxy with a velocity of about 230 km/sec, completing a revolution in about 190 million years. The total number of stars in the galaxy is tremendous, $120 \cdot 10^9$, and the total mass of the galaxy is $15{,}864 \cdot 10^{43}$ g, which is $8 \cdot 10^{10}$ times the mass of the sun.

Powerful sources of radio emission have been detected in the galaxy. Three types of them are known now. The maximum radio emission of the first type is situated near the center of the galaxy. Its source is clouds of interstellar gas, ionized by the light of the nearest hot stars. The second type of radio emission is associated with the energy radiated by free electrons moving with tremendous velocities in the weak magnetic fields of the rarefied medium between the interstellar gas clouds. The first and second sources usually have large angular dimensions (up to 20' and more) and are called radio nebulae. Radio emission sources of the third type, in contrast to the first two, have small angular dimensions (1 - 10'), and therefore they are sometimes called "radio stars."

Some "radio nebulae" are formed by the explosion of supernovas; the Crab nebulae in the constellation Taurus (Figure 5) consists of the remnants of such an explosion. However, most of the presently known "radio stars" and "radio nebulae" are not connected with our galaxy, but they are situated beyond its boundaries.

Our galaxy is made of substances which consist of atoms whose nuclei contain protons and neutrons and whose outer shells contain electrons. Some scientists assume that there are galxies in the universe which consist of antimatter — atoms having antiprotons in the nucleus and positrons in the shell. In these worlds, antiparticles are stable and our particles, unstable. However, in spite of this, all the physicochemical properties of the atoms in both worlds would be identical. In "anti-worlds" there would be the same chemical compounds with the same composition and properties, and it is perfectly possible that there exist in them the same organic and inorganic materials, and perhaps the same living organisms, and even human beings, as in our world.

The sun is the star nearest to us. It is designated by the symbol ⊙. The mean distance from the earth to the sun is equal to one astronomical unit, i.e., 149,457,000 km.* This distance is traversed by a light ray in 8 min 18 sec.

Due to the eccentricity of the terrestrial orbit, the distance from the earth to the sun varies approximately within ± 5 million km. The diameter of the sun is 1,390,000 km, which is 109.1 times as large as the diameter of the earth, and the angular diameter of the solar disk as seen from the earth is about 32'.

The mass of the sun is $1.99 \cdot 10^{33}$ g, which amounts to 99.86% of the mass of all the bodies of the solar system. The average density of the sun is not high, being equal to 1.41 g/cm^3, and the gravity acceleration on its surface is 275 m/sec^2, which is approximately 28 times as much as that on the earth. The circular velocity for the surface of the sun** is 439.3 km/sec and the escape velocity† is 619.4 km/sec.

The sun is an incandescent gaseous body. The temperature of its surface layers is about 6000°C. In the center of the sun the temperature reaches 20,000,000°C. The sun is the source of a tremendous amount of energy. Each second it sends to the earth 40 quintillion kilocalories of heat. A considerable part of this energy is scattered and partially absorbed by the atmosphere. On the average 30% of this solar energy reaches the earth's surface in the course of one year. This energy would be sufficient to melt and boil a solid layer of ice 1000 km thick around the earth.

According to calculations, the sun loses about 240 million tons of its mass every minute through radiation.

Main composition of the thermal radiation of the sun

	%
Infrared rays	51.0
Visible rays	41.0
Ultraviolet rays	7.7

Along with the the thermal radiation, the sun sends into space fluxes of charged particles of matter††. These particles, accelerated in the magnetic fields of the sun, acquire tremendous energy – the energy of cosmic rays.

Relative composition of the cosmic radiation

	%
Protons	80
α-particles	19
C, N, O	0.66
Na, Mg, Al, Si	0.12
S, A, Ca	0.04
Fe	0.02

The maximum intensity of the cosmic ray fluxes reaches 10^9 particles per cm^2/sec.

* [Notice that a few pages earlier, the value of the a.u. is given as 149,599,300 km.]

** [The circular velocity is the velocity of a theoretical satellite whose orbit is just above the surface of the sun. Cosmic velocities are dealt with in Chapter II.]

† [Escape velocity is the velocity required of a body to leave the gravitational field of the celestial body from which it is launched.]

†† So-called corpuscular radiation.

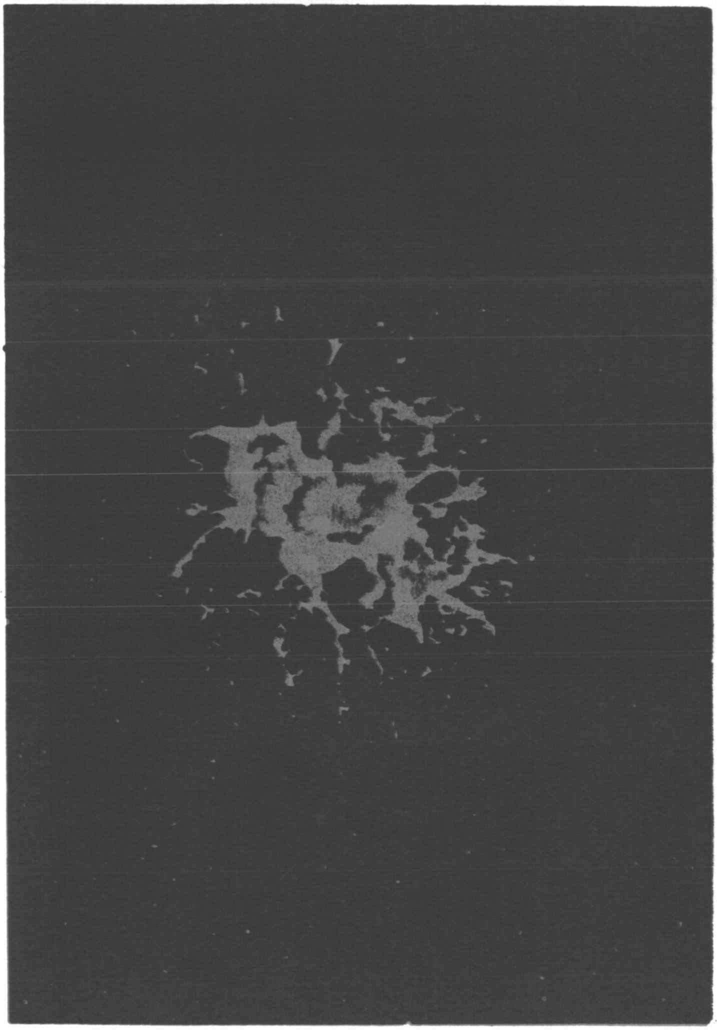

FIGURE 5. Photograph of the Crab nebula in the constellation Taurus, obtained through a light filter which transmits only the red line of nitrogen

The tremendous energy production of the sun is due to thermonuclear reactions taking place in the interior of the sun as a result of the high temperature and huge pressure existing in its center.

The visible surface of the sun is called the photosphere. The photosphere is the conventionally accepted boundary of the solar gaseous sphere, over which the solar atmosphere is situated. Sunspots, faculae, and grains are observed on the surface of the sun (Figure 6).

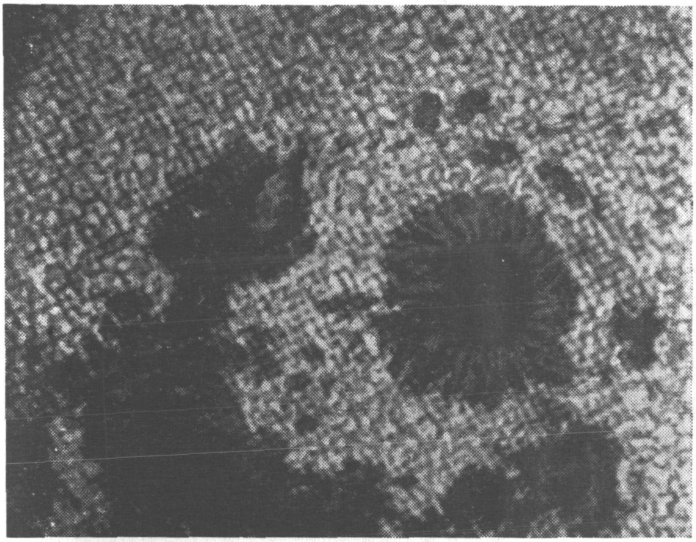

FIGURE 6. Sunspots and grains

Sunspots are solar formations with a temperature of 4000 to 4500 °C. The spots look dark solely by reason of the contrast with the brighter photosphere, which has a higher temperature. The spots have diverse forms, but are mostly circular. A sunspot is a vortex of turbulent gases on the sun. The lifespan of a spot is from one day to several months. The dimensions of sunspots often reach 90,000 km in diameter, which is approximately 7 times the diameter of the earth. Occasionally, sunspots of considerably larger dimensions appear. Thus, in March 1947, a spot of 214,600 km was observed. Often pairs and groups of spots are formed on the surface of the sun.

The spots move from the eastern to the western limb of the sun due to the rotation of the sun about its axis; they also move independently over the surface of the sun.

The solar activity depends on the number of spots and the relative area of the sun covered by them. Usually spots appear periodically on the sun. The complete period of variation of the solar activity is approximately 22 years. Maximum solar activity is observed on the average every 11.11 years. The last maximum occurred in 1958.

Faculae on the solar surface have the form of hairlike filaments of different forms, whose brightness is higher than the brightness of the photosphere. Sunspots are always accompanied by faculae, but on occasions faculae are also observed separately from sunspots. Sometimes facular fields are formed which cover considerable sections of the solar surface.

Grains are bright formations of oval form, covering the whole photosphere in a reticular pattern. The dimensions of grains are relatively small, reaching 400 to 500 km. Their temperature is 100 to 200°C higher than the temperature of the photosphere. The lifespan of an individual grain is only several minutes.

The sun is surrounded by an incandescent luminous atmosphere. A thin layer of the solar atmosphere (about 500 km thick) lying immediately over the photosphere is called the reversing layer. It passes gradually into the chromosphere, visible during total solar eclipses as a thin reddish ring around the sun. The chromosphere reaches a height of 15,000 km.

Calcium, hydrogen, helium, iron, titanium, and other chemical elements have been observed in the solar atmosphere. The lowest layer of the solar atmosphere has a gas pressure a thousand times lower than the atmospheric pressure at the surface of the earth.

Very bright, short-lived flares — eruptions of gas that flare up rapidly — are observed in the chromosphere. The lifespan of these flares is only several minutes. They most often appear during the development or decay of sunspots. Flares are peculiar explosions, resulting from a rapid compression of magnetic fields, leading to a short-lived heating of a small volume of solar gas to a temperature of about 30 million degrees. According to the estimate of A. Sebernyi, corresponding member of the Academy of Sciences of the USSR, some solar flares are equivalent to a simultaneous explosion of from 30 to 100 thousand megaton atomic bombs over the area of the flare. New interesting data on solar flares have been obtained by means of unmanned interplanetary stations and space rockets.

During solar flares, short-wave, X-ray, and ultraviolet radiation is generated, and protons with an energy up to 100 million ev are emitted. The short-wave radiations enhance the ionization of the earth's ionosphere which sometimes results in interruption of radio communication in the short-wave range. Penetration of protons into the earth's atmosphere causes absorption of radio waves in the polar regions. These protons also constitute a serious hazard for space flights.

In addition to these, large particle fluxes, moving with velocities of 1000 km/sec and more, are ejected by flares. They cause the aurora polaris and magnetic storms on the earth. Finally, flares are accompanied by powerful radio-wave emissions, which sometimes lead to a disturbance in the operation of radar devices and a loss of radar visibility of the target.

With very strong flares, the flow of cosmic rays moving in the direction of the earth sharply increases, indicating the formation of particles of a high energy — up to 100 billion ev.

Cosmic radiation constitutes a serious danger both for the crew and equipment of spacecraft. It causes a decrease in the insulation properties of insulating materials, a modification of the properties of plastic materials, and possibly even a modification in the properties of metals. Plastic materials are particularly sensitive to cosmic radiation. Failures of electronic equipment are also possible.

According to calculations, the weight of the radiation shield of a spaceship should amount on the average to about a quarter of a ton per square meter of the shielded area /22/. Shielding from radiation danger in space flights is a serious problem which scientists and engineers must solve.

In various places, huge jets or flame tongues are ejected from the chromosphere and ascend tens and hundreds of thousands of kilometers. They are called prominences (Figure 7). The formation of prominences is a periodical process, coinciding mainly with solar activity, but the periodicity here is less clearly defined and, also, the prominence maxima occur usually earlier than the peak of solar activity.

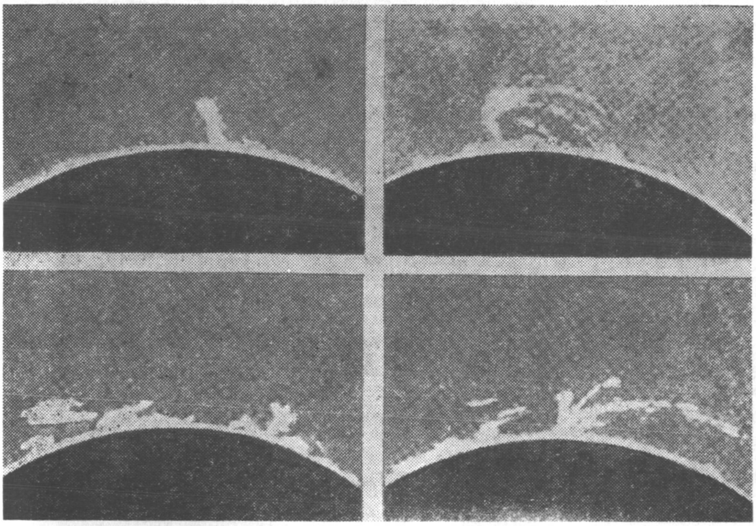

FIGURE 7. Different types of prominences

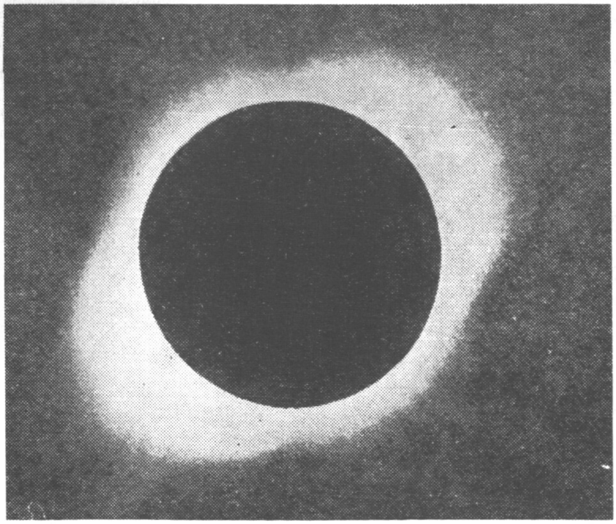

FIGURE 8. Solar corona during the eclipse of 30 June, 1954

The broad brilliant halo observed during total solar eclipses around the sun is called the solar corona (Figure 8). The lower boundary of the corona is the chromosphere. The corona extends over distances considerably beyond the sun's radius. The further from the sun, the weaker it gets, gradually merging with the background of the sky. In several instances, the corona has been observed extending to distances of 4 to 5 solar radii from the surface of the sun.

The form of the corona is different in different years and changes in accordance with the amount of sunspots and prominences. At periods of maximum solar activity, the corona surrounds the sun on all sides approximately equally; at minimum it is elongated at the solar equator, and at the poles it reduces to short rays. This leads to the conclusion that the reasons for the formation of the corona are closely related to the processes which take place on the surface and in the atmosphere of the sun.

In 1942 — 1945 the radio emission of the sun was detected in quite a wide band (8 mm to 15 m). The radiation of the chromosphere has wavelengths of the order of centimeters, the radiation of the corona, of the order of meters. The radio emission of the sun is due to the solar atmosphere, which, as any strongly heated body, is a source of electromagnetic energy. The intensity of radio emission does not remain constant. Individual surges in the radio emission are connected with sunspots and chromospheric flares.

For a systematic study of all the phenomena of the sun and for a forecasting of the geophysical phenomena caused by the variation of the solar activity, the so-called "Sun Service" was organized on an international scale. A whole series of observatories in many countries, including the USSR, conduct daily observations according to one plan. In the USSR all the observations are sent for processing to the central "Sun Service," headed by the Solar Research Committee of the Academy of Sciences of the USSR. Particularly valuable results on observation of the sun and in regard to clarifying the connection between geophysical phenomena and solar activity were obtained during the International Geophysical Year of 1957 - 1958, which was chosen to coincide with the time of maximum solar activity.

The sun exerts a great influence on the conditions of space flights. The powerful attraction, or gravitational field of the sun, is a factor determining the choice of the trajectory of the spaceship, the required velocities for the execution of the interplanetary flight, and its periods. The flux of charged particles and the sun's intense ultraviolet radiation necessitate the forecasting of solar activity, and choosing of times most favorable for space flights; this involves observation of solar activity, and the organization of a warning service against excessive radiation intensity. It is also necessary to take into account in space flight the variations of the magnetic field of the earth and disturbances of radio reception connected with solar activity. Finally, the sun, like any other celestial body, can be used for the determination of the motion parameters of a spaceship.

Much is still unclear as to the nature of the sun, its influence on geophysical processes, and the conditions of space flights. Unprecedented possibilities for the solution of these problems are opened by the launching of artificial earth satellites, automatic interplanetary stations, and spaceships.

The sun is the central body of the solar system (Figure 9), which includes the major planets and their satellites, the minor planets, or the asteroids, comets, meteor showers and individual meteors, dustlike matter, and also some meteoric material scattered in interplanetary space.

Nine major planets are known at present: Mercury, Venus, Earth, Mars, Jupiter, Saturn, Uranus, Neptune, Pluto, as well as 31 satellites of these planets. The number of minor planets discovered, moving mainly between the orbits of Mars and Jupiter, amounts to over 1800 and the number of comets to about 500. It is assumed that there are over 100,000 comets and 50,000 to 100,000 minor planets in the solar system. It is also possible that there are other major planets situated beyond the orbit of Pluto. There is already strong evidence of the existence of a tenth major planet in the solar system — Transpluto. The distance of this planet from the sun was calculated and its revolution period about the sun determined, but we have not yet succeeded in observing it directly.

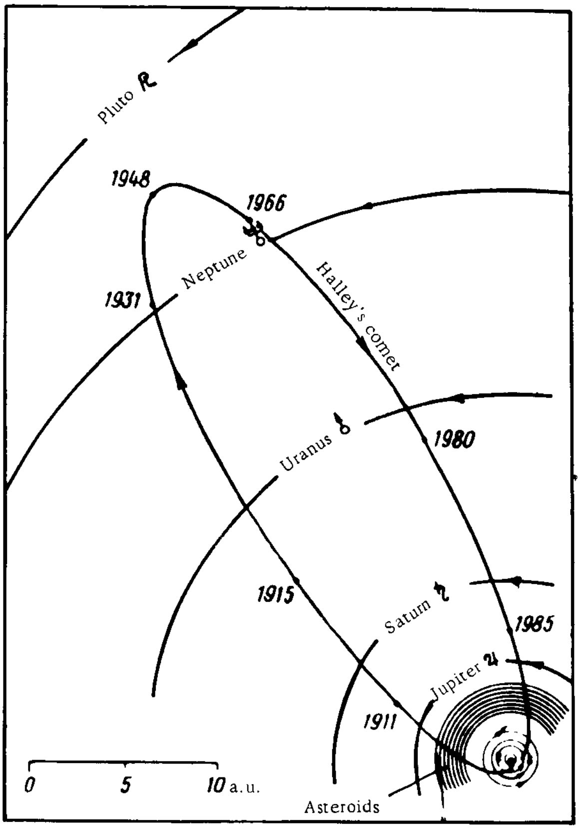

FIGURE 9. The solar system with the orbit of Halley's comet (the position of the comet in individual years is indicated on the orbit)

Compared with the sun, the dimensions of the planets are small (Figure 10). Their orbits lie approximately in the same plane, the plane of the ecliptic (Figure 11).

All the bodies of the solar system revolve, strictly speaking, not around the sun, but around the common center of gravity of the whole solar system, with respect to which the sun itself describes a very complicated curve. However, since most of the mass of the entire system is concentrated in the sun, the displacement of the center of revolution is small.

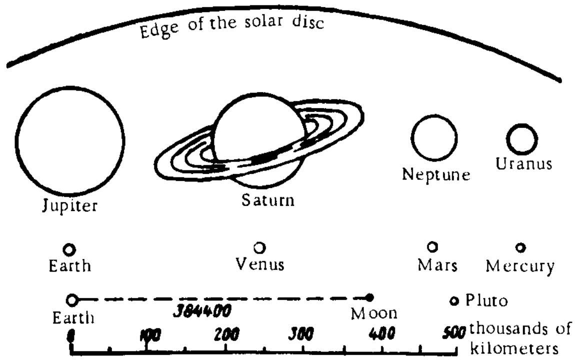

FIGURE 10. Comparative dimensions of the major planets and of the sun

FIGURE 11. Inclination of planetary orbits to the plane of the Earth's orbit (the plane of the ecliptic)

What are the boundaries of the solar system? They are determined by the distances of the remotest satellites of the sun — the so-called long-period comets with periods of revolution close to one and a half million years. Some of these comets at maximum distance are 6 trillion km ($6 \cdot 10^{12}$ km) from the sun. This distance may be considered as the boundary of the solar system.

The sun together with all its satellites is moving in space relative to its surrounding stars with a velocity of about 19.5 km/sec towards the constellation Hercules. As has already been said, the solar system participates in the revolution of the entire galaxy around the galactic center.

We have already mentioned the fantastically great distances to the celestial bodies. Their remoteness puts in doubt the possibility of flights to other stars and, for the present and the near future, space flights will have to be limited to the solar system, and more specifically, to the nearest bodies of the solar system, i.e., the moon, Venus, Mars, and the nearest minor planets. Therefore astronauts and pilots of spaceships should have a sound knowledge of the physical characteristics of the bodies of the solar system, the laws of their motion, and the conditions which they will encounter on the moon and on the nearest planets. Before considering these problems it is necessary to study, at least briefly, the various astronomical coordinate systems and also the elements which determine the motion parameters of celestial bodies in space. Otherwise, it is not possible to characterize the bodies of the solar system from the point of view of the requirements of areonautics and space navigation.

§3. Systems of Celestial Coordinates. Elements of Planetary Orbits

Several coordinate systems are used in astronomy to determine the position of a celestial body in space. The galactic coordinate system is convenient for determining the position of stars and star clusters of our galaxy, as well as of star systems situated beyond the limits of the galaxy. The positions of planets with respect to the sun and to the earth's orbit are more conveniently determined in the so-called ecliptic coordinate system, whereas the positions of celestial bodies with respect to the earth are most conveniently determined in the equator and horizon systems of celestial coordinates. The names of these coordinate systems originate from the names of the planes which are basic in the given system. Naturally only the last three coordinate systems are of interest to us and these will therefore be discussed in more detail.

In any system of spherical celestial coordinates, use is made of an auxiliary sphere of arbitrary radius, called the celestial sphere, with the center at one point in space. The celestial body is then projected on this sphere.

The ecliptic system of coordinates. The ecliptic is the well-known term applied to the great circle of the celestial sphere formed by the intersection of the plane of the earth's orbit with the celestial (star) sphere. The annual motion of the sun among the stars, as seen from the earth, takes place along the ecliptic.

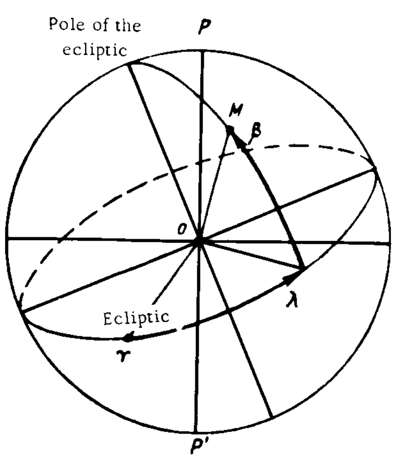

FIGURE 12. Ecliptic system of coordinates: P and P' - poles of the earth; ϒ- vernal equinox; β and λ - astronomical latitude and longitude of the celestial body M

In the ecliptic coordinate system, the center of the celestial sphere is made to coincide with either the center of the sun, in which case the ecliptic coordinates of the heavenly body are called heliocentric, or with the center of the earth, in which case, the coordinates are called geocentric.

The principal circles of the ecliptic coordinate system are the ecliptic and the latitude circle of the celestial body (Figure 12). The latitude circle of a celestial body is the term applied to the great circle of the celestial sphere passing through the body and through the poles of the ecliptic. The position of the body in this coordinate system is determined by the astronomical longitude λ, and the astronomical latitude β. The astronomical latitude of a celestial body is measured from the vernal equinox or Aries (ϒ) (the point of the ecliptic at which the sun is situated on 21 March of each year) along the ecliptic in the direction of the annual motion of the sun to the point of intersection of the latitude circle of the body with the ecliptic. The astronomical latitude is measured along the latitude circle of the body in both directions from the ecliptic from 0 to ± 90°.

In the equator system of coordinates (Figure 13), the basic plane is the plane of the celestial equator, coinciding with the plane of the terrestrial equator. The continuation of the axis of revolution of the earth formes the axis of the celestial sphere, and its intersection with the celestial sphere forms the sphere's northern (P) and southern (P') poles. The great circle of the celestial sphere passing through these poles is called the hour circle of the celestial body. The ecliptic intersects the equator at two points, the vernal (♈) and the autumnal (♎) equinoxes, and the vernal equinox is therefore common to both the ecliptic and the celestial equator.

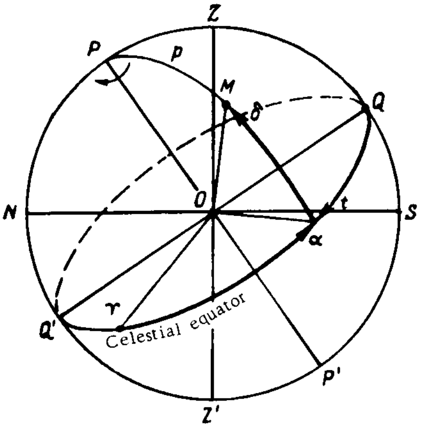

FIGURE 13. Equator system of coordinates

Z and Z' - zenith and nadir; α and δ - right ascension and declination of the celestial body M; t - hour angle; p - polar distance

In this coordinate system the position of a celestial body on the celestial sphere is determined by the right ascension α and the declination δ of the body. The right ascension is measured from the vernal equinox along the celestial equator in a direction opposite to the apparent daily rotation of the celestial sphere up to the point of intersection of the hour circle of the body with the celestial equator. Sometimes the right ascension is expressed not in degrees, but in time units, taking 360° as corresponding to 24 hours. The declination of a celestial body is measured along the hour circle of the body in both directions from the celestial equation from 0 to ± 90°.

The apparent rotation of the celestial sphere resulting from the daily rotation of the earth relates to the same axis and this means that the equatorial coordinates of celestial bodies do not vary. This makes it possible to construct in this coordinate system maps of the star sky and star atlases.

Sometimes the position of the celestial body is determined by the declination of the body and its hour angle. The great circle of the celestial sphere which passes through the zenith and the poles is called the meridian of the observer. The angle between the meridian of the observer and the hour circle of the celestial body is called the hour angle t of the body. The hour angle is measured from the point Q of the equator in the direction of rotation of the celestial sphere. In conclusion it should be noted that the hour angle varies in a uniform manner: in 24 hours it varies by 360°.

Knowing the equatorial coordinates of a spaceship makes it possible to determine its position among the stars, e.g., for purposes of optical tracking. Thus, in one of the TASS communiques on the 1961 launching of the automatic interplanetary station to the planet Venus, the precalculated equatorial coordinates of the station for 1200 hours Moscow time for 3 March, 1961, were given. The announcement said: "The right ascension of the automatic interplanetary station at this time will be 0 hours 21 minutes 31 seconds: the declination, minus 1 degree 03 seconds."

The horizon system of coordinates (Figure 14), whose basic plane is the plane of the true horizon, is used to determine the position of a celestial body with respect to the earth's surface. Knowing the horizontal coordinates of a spacecraft makes it possible to estimate the conditions for observation at a given time from a given point on the earth. The pole of this coordinate system is the piont of intersection of the plumb line with the celestial sphere, called the zenith z. The great circle of the celestial sphere passing through the zenith, through the opposite point (nadir z') and through the celestial body is called the vertical of the body.

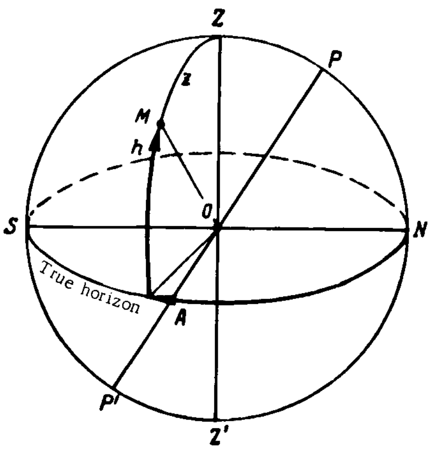

FIGURE 14. Horizon system of coordinates

A and h - the azimuth and altitude of the celestial body M; z - the zenith distance of the body.

The position of a celestial body and of a cosmic object in the horizontal coordinate system is determined by the azimuth A and altitude h. The azimuth is measured by the arc of the true horizon from the north point N to the point of intersection of the vertical with the true horizon. The azimuth is measured from the north point to the east. The altitude is measured by the arc of the vertical from the true horizon to the celestial body within the range of 0 to ± 90°. Celestial bodies and spaceships having negative altitudes are situated below the horizon and are not visible from the given point on the earth.

If we project the celestial body on the surface of the earth, we obtain a point at which its altitude is equal to 90°. This point is called the geographical place of the body, G. P. Owing to the rotation of the earth, this point moves over the earth's surface. By analogy, the point at which the height of the spaceship is equal to 90° can be called the geographical place of the spaceship. To define the position of a spacecraft as viewed from the earth, this point can be calculated for a given moment. Thus, in the above-mentioned TASS announcement, it was said: "By 1200 hours Moscow time on 3 March, 1961, the station will be over the point with the coordinates 1 degree 15 minutes south latitude and 69 degrees 30 minutes east longitude." These were the coordinates of the geographical place of the interplanetary station at the indicated moment.

Transformation from one coordinate system to another is made using the formulas of spherical trigonometry.

The above systems of spherical coordinates have a serious disadvantage; they do not allow for the determination of the position of the celestial body in space, but only the position of the body on a sphere of arbitrary radius.

This, however, is completely sufficient to solve most problems of practical astronomy. Indeed, the airplane pilot who is measuring the coordinates of celestial bodies in order to determine the position of the airplane is not interested in the distances to them; he only needs to know the angular coordinates of the celestial bodies, which determine their position on the celestial sphere. However, in order to solve most of the problems of space navigation one has to know the position in space of the spacecraft and of the celestial bodies.

Thus, for flights inside the solar system, it is possible to use a rectangular coordinate system commencing at the center of the sun (Figure 15). The OX axis of this coordinate system can be directed to the vernal equinox Υ, the OY axis lies in the plane of the ecliptic, and the OZ axis is perpendicular to the OX and OY axes and directed to the pole of the ecliptic. The rectangular coordinates x, y, and z of a spaceship give an unequivocal determination of its position in space.

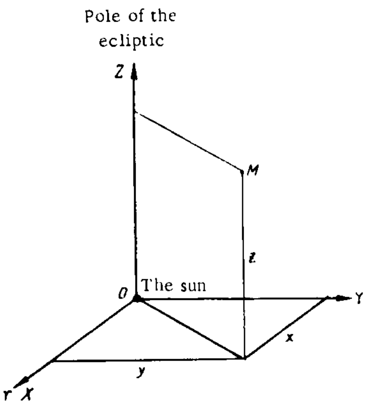

FIGURE 15. Rectangular heliocentric coordinate system for the solution of problems of space navigation

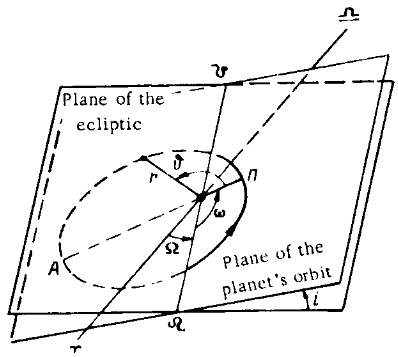

FIGURE 16. Elements of planetary orbits

A - aphelion; Π - perihelion; i - inclination of the orbit; r - radius-vector of the planet; Ω - right ascension of the ascending node; ω - angular distance of the perihelion from the node; ϑ - the true anomaly of the planet

It is, of course, possible to choose other coordinate systems. For example, the positions of a spacecraft and of celestial bodies can be determined in a rectangular coordinate system commencing at the center of the earth and with the plane XOY coinciding with the equatorial plane of the earth.

The calculation of the current spatial coordinates of celestial bodies and spaceships is possible only if the elements of their orbits are known. The motion of planets with respect to the sun takes place along elliptic orbits.

The position of the orbit of a planet in space and of the planet on its orbit are determined by the elements of the planet's orbit; namely, the inclination of the orbit, longitude of the ascending node of the orbit, distance of the perihelion from the node, semimajor axis, eccentricity, and mean anomaly at a given epoch (at a given moment) or the moment of passage through the perihelion (Figure 16).

The orbit inclination i is the angle between the plane of the ecliptic and the orbital plane of a planet. It may vary from 0 to 180°. An angle i larger than 90° corresponds to a so-called reverse motion of the celestial body.

The ascending node ☊ of the orbit is the term applied to the intersection point of the orbit with the plane of the ecliptic at which the planet passes from the southern to the northern hemisphere of the ecliptic. The opposite point of the orbit is called descending node ☋, and the line connecting these two points is called the line of nodes. The line of nodes lies both in the plane of the orbit and in the plane of the ecliptic.

The longitude of the ascending node of the orbit Ω is the angle in the plane of the ecliptic which is enclosed between the straight line connecting the vernal equinox with the center of the sun and the straight line connecting the ascending node with the center of the sun.

The point of the orbit nearest to the sun is called perihelion (Π), and the farthest point is the aphelion (A). The line connecting these two points is called the line of apsides. The distance ω of the perihelion from the node of the orbit is measured by the angle between the line of nodes and the line of apsides. It is measured off along the orbit in the direction of motion of the planet from the ascending node to the perihelion and determines the orientation of the orbit in its plane. Sometimes the orientation of the orbit is determined by the longitude of the perihelion $\Omega' = \Omega + \omega$. Consequently the angle Ω' is measured in two planes: in the plane of the ecliptic up to the line of nodes and in the plane of the orbit from the line of nodes to the line of apsides. These elements determine the position of the planet's orbit in space.

The semimajor axis, a, of the orbit, is equal to half the distance from the perihelion to the aphelion. The eccentricity, e, of the orbit determines the geometrical form of the orbit and is given by the formula

$$e = \frac{c}{a} = \frac{\sqrt{a^2 - b^2}}{a},$$

where c is the distance from the geometrical center of the elliptic orbit to its focus (the center of the sun) and b is the semiminor axis of the planet's orbit.

For elliptic orbits $0 < e < 1$. In the case of eccentricities close to zero, the form of the orbit is close to a circle.

The mean anomaly M is a quantity equal at any moment to the arc which the planet would describe after passing through the perihelion if moved uniformly in a circular orbit completing one revolution in the period of rotation P of the planet. Obviously, the mean diurnal motion of the planet is $360°/P$.

Let us now denote by T_0 the time of the passage of the planet through the perihelion. Then, for the time T the mean anomaly is

$$M = \frac{360°}{P}(T - T_0).$$

Together with the mean anomaly the true anomaly of the planet has also to be considered.

The true anomaly ϑ is the term applied to the angle which is formed by the line of apsides and the straight line r connecting the center of the sun and the planet. This angle is measured from the perihelion in the direction of motion of the planet. The true anomaly determines the position of the planet on the orbit.

The motion of the planet along the orbit is characterized by its period of revolution. We distinguish between sidereal and synodic periods of revolution. The sidereal or stellar period of revolution is the term applied to the time interval in which an observer situated on the sun would see the planet in the former position with respect to the stars. The synodic period is the term applied to the time interval in which the planet, as viewed by a terrestrial observer, arrives at its former position with respect to the sun. It is perfectly clear that owing to the earth's motion along its orbit around the sun these two periods are different. Thus, the sidereal period of revolution of the planet Mars is 1 year 231.73 days, whereas the synodical period is almost 2 years 50 days.

These are the elements which determine the orientation of the orbit of a planet in space and the position of the planet on the orbit. It should be noted that these same elements also characterize the orbits of interplanetary ships and the orbits of artificial satellites of planets. The only difference is that the inclination of the orbit of satellites is measured with respect to the plane of the planet's equator, and the designation of the extreme points of the orbit includes a term denoting the planet.*

For artificial satellites of the earth these points are called perigee and apogee**; of Venus, peri- and apovenus; of Mars, peri- and apomars; of Saturn, peri- and aposaturn; of Jupiter, peri- and apojovian.† The extreme points of the orbits of artifical satellites of the moon are called peri- and apomoon or peri- and aposelena.††

§4. The Earth

For spacecraft of any destination the earth constitutes the planet of departure and arrival.

The earth is an ordinary planet of the solar system. Its astronomical designation is ♁ or ⊕. The true form of the earth — geoid — is close to an ellipsoid of revolution having a small flattening at the poles. The terrestrial ellipsoid is now thought to have the following characteristics:

	International ellipsoid, 1910	Soviet ellipsoid, 1940
Equatorial radius, km	6378.388	6378.245
Polar radius, km	6356.912	6356.863
Flattening	1 : 297.0	1 : 298.3

* The extreme points of the orbits of spaceships are sometimes called pericenter and apocenter.

** From the Greek word geo — Earth.

† From the Latin name of Jupiter — Jove.

†† The ancient name of the moon was Selena.

For solving practical problems not requiring a high degree of accuracy, the earth is taken to be a sphere of radius 6371 km.

Here are some physical characteristics of the earth. The area of its surface is 510,083,000 km^2, its volume is $1.083320 \cdot 10^{12}$ km^3 and its mass is $5.974 \cdot 10^{27}$ g. About 7% of the mass of the earth is made up of water in its various forms. The average density of the earth is 5.517 g/cm^3, and the average density of the surface layers is only 2.65 g/cm^3. The circular velocity and escape velocity for the surface of the earth are 7.9 km/sec and 11.2 km/sec, respectively.

The mean distance of the earth from the sun is 149,457,000 km. The earth moves around the sun along an elliptical orbit with a mean velocity of 29.76 km/sec, or about 100,000 km/hr.

The sidereal revolution period of the earth is 1 year 0.006 days. The inclination of the earth's equator to its orbit is 23°27'. The eccentricity of earth's orbit is small, being equal to 0.01673. This indicates that the orbit of the earth is very close to a circle. The perihelion of its orbit occurs on about 3 January and the aphelion on 4 July.

Owing to the diurnal rotation of the earth, all the points of its surface have a certain linear velocity V_d, which can be determined by the formula (in m/sec):

$$V_d = 465 \cos \varphi,$$

where φ is the latitude of the given point.

This velocity, naturally, should be taken into account in the launching of spacecraft. When a spacecraft is launched in an eastern direction this velocity "helps" it to get into orbit. If the spacecraft is launched, for example, from the equator to the east and the required velocity for entering the appropriate orbit is 7.465 km/sec, then the rocket engines need only give the ship a velocity of 7.0 km/sec. The remaining 0.465 km/sec are "added" to the ship by the rotating earth.

The most important physical characteristic of the earth is the gravitational acceleration on its surface. The gravitational acceleration determines the form and parameters of the orbits of spaceships leaving the earth, and for a given orbit it determines the required initial velocity.

Theoretical and experimental studies show that the gravitational acceleration is not constant for different points of the earth's surface. This fact was first noted in 1672 by the French astronomer Riche. Fifteen years later Isaac Newton proved theoretically that due to the flattening of the earth as a result of its diurnal rotation the gravitational acceleration should decrease from the poles to the equator. This conclusion was subsequently confirmed by numerous direct measurements at various points of the earth's surface.

Therefore the gravitational acceleration reduced to sea level depends on the average only on the latitude of the location. Its magnitude g_φ for any point of the earth's surface can be calculated by the formula

$$g_\varphi = g_0 + (g_{90^\circ} - g_0) \sin^2 \varphi,$$

where g_0 and g_{90° are the gravity accelerations at the equator and at the poles, respectively. For the pole g_{90° = 983.2 cm/sec^2, and for the equator g_0 = 978.0 cm/sec^2.

For practical calculations not requiring a high degree of accuracy, the gravitational acceleration for the whole earth is equal to 981 cm/sec^2.

The results of direct measurements of the gravitational acceleration at various points of the earth's surface differ in a number of cases from calculated values. These deviations are called anomalies of the gravitational force.

The anomalies are due to nonuniformity in the structure of the earth's core, both with respect to visible external masses (mountain masses, islands, and so on), and to the density of the rocks forming the core. However, the magnitude and character of the anomalies are also affected by the internal layers of the earth.

In addition to the gravitational field, a magnetic field has also been observed in the space around the earth. At present, it is assumed that the general magnetic field of the earth can be represented as a sum of many magnetic fields: the field of uniform magnetization of the terrestrial sphere, originating mainly from the internal layers of the earth; the continental or residual field, due to the nonuniform structure of the internal layers of the terrestrial core; the anomalous field, due to magnetization of the upper layers of the terrestrial core; the external field, produced by electric currents in the upper layers of the atmosphere; finally, the variation field, due to periodic variations in the character of the motion of charged particles in the upper layers of the atmosphere.

For a long time, the magnetic field of the earth has helped to solve a number of the most important problems of ship and airplane navigation. Ordinary magnetic, gyromagnetic, and gyroinduction compasses are used to measure the vehicle's course and to perform the navigation in a given direction. Measurements of the magnitude of the elements of the terrestrial magnetism, for example, the total force of the magnetism or its components, allow the navigator to obtain the position line of the airplane. Terrestrial magnetism will also aid space navigation. For example, many scientists are considering the possibility of stabilizing the orbits of low-orbit satellites by means of a sensitive magnetic element.

The motion of spaceships in the earth's magnetic field causes some dynamic effects, which result in a deviation of the actual orbit of the spaceship from the calculated one. These deviations, though small, should be taken into account in a number of cases.

A quantitative estimate of terrestrial magnetism can be given by the magnetic moment of the earth, which is the product of the volume of the earth and its magnetization. The magnetic moment of the earth is equal to $8.3 \cdot 10^{25}$ cgsm units, or $8.3 \cdot 10^{15}$ weber · m.* Such a magnetic moment is equal to that of a sphere of nickel-aluminum alloy (alnico) with a radius of about 500 km which is magnetized to the maximum. This example gives some idea of the degree of magnetization of the earth. The magnetic poles of the earth do not coincide with the geographical poles. The coordinates of the north magnetic pole are $\varphi = 74°$ north latitude, $\lambda = 101°$ west longitude; and the south magnetic pole, $\varphi = 69°$ south latitude, $\lambda = 143°$ east longitude.

The degree of magnetization of the earth can also be estimated by the distribution of the magnetic field intensity at different points of the earth's

* A weber is the magnetic flux which, when linked with a single turn, generates an electromotive force of 1 volt in the turn, as it decreases uniformly to zero in one second.

surface. The magnetic field intensity at the magnetic equator is 0.35 oersteds, and at the magnetic pole, 0.65 oersteds.*

Moving north along a magnetic meridian a distance of one km at middle latitudes, the vertical component Z increases, and the horizontal component of the magnetic field of the earth H decreases approximately by 3 to 5 γ. At a height of 20 km, the horizontal component is lower than its value at the earth's surface by only 1%, and at a height of 200 km, by 10%. On the average, the horizontal component decreases by 7 γ for each kilometer of height. With increasing height, the magnitude of the vertical component of the magnetic field of the earth decreases also.

The magnetic field of the earth extends to 100,000 km for all practical purposes.

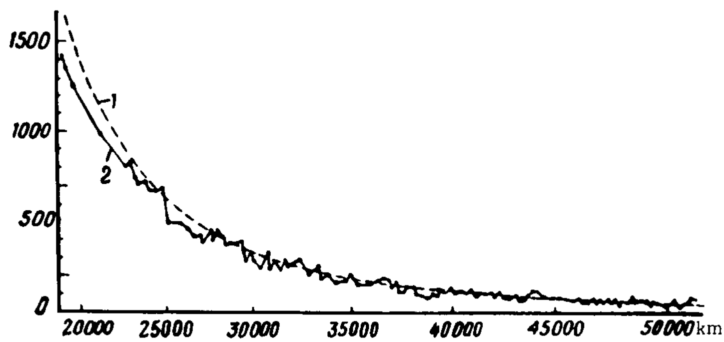

FIGURE 17. Intensity of the earth's magnetic field as a function of the distance from its center:

1- calculated variation of the magnetic field intensity; 2- intensity, measured by the second rocket launched in the direction of the moon.

Instruments mounted on the first satellites detected a considerable difference between the real magnetic field of the earth and the calculated one (Figure 17). This is apparently due to strong electric currents tens of thousands of kilometers from the earth. The difference is particularly large at distances of 20,000 to 22,000 km from the center of the earth. No satisfactory theory of the earth's magnetism has yet been formulated. The world magnetic survey during 1964 - 1965 will result in more accurate data on the distribution of the elements of the magnetic field of the earth over its surface and in space.

The earth is surrounded by a gaseous envelope, the atmosphere. The motion of spaceships starting from the earth takes place in the atmosphere, low-orbit spaceships move in the upper layers of the atmosphere, and the paths of ships returning from space pass through the atmosphere. Overcoming the resistance of the earth's atmosphere requires additional energy expenditures to put a spaceship into orbit. The earth's atmosphere causes a considerable heating of the ship and tremendous overloads on a descending spaceship. The following is a brief description of the earth's atmosphere.

* A field intensity of 1 oersted acts on a unit positive magnetic mass with a force equal to 1 dyne. A smaller intensity unit is the gamma, γ, equal to 0.00001 oersted.

The atmosphere participates in the rotational motion of the earth about its axis. In addition, it has a complicated motion with respect to the surface of the earth, which results in a continual change of its physical state.

The atmosphere is a mechanical mixture of gases, for the most part nitrogen, oxygen, and argon. In addition, it contains varying amounts of water vapor and carbon dioxide, and negligible amounts of hydrogen, helium, neon, xenon, krypton, ozone, as well as methane and oxides of nitrogen. The main gases constitute 99.97%, and the remaining gases only about 0.03% of the atmosphere.

The following figures give some idea of the amount of gases in the earth's atmosphere: the total weight of the atmosphere is approximately $5 \cdot 10^{15}$ ton; if the entire atmosphere could be compressed to the density of water, then the globe would be covered by a 10 m thick uniform layer of compressed atmospheric gases. It has been found that the relative content of the main gases remains practically constant up to heights of 100 to 120 km. The absolute content of all the gases decreases with height. The atmosphere is nonuniform in the vertical direction also with respect to its other physical parameters: temperature, pressure, and so on (Figure 18).

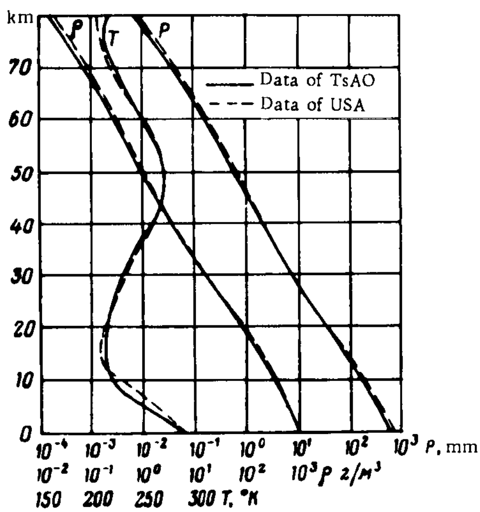

FIGURE 18. Vertical distribution of the pressure P, temperature T, and density ρ according to measurements from rockets made by the Central Aerological Observatory of the USSR (TsAO) and from those made in the USA

Depending on the variation of its physical properties, the atmosphere is divided into several layers, or spheres (Table 1).

The troposphere is the atmospheric layer whose physical properties are mostly dependent on the earth's surface. A characteristic feature of the troposphere is the drop of air temperature with increasing height, reaching on the average 0.65°C per 100 m of height. All the meteorological phenomena are observed in the troposphere.

The highest wind velocities are observed in the upper troposphere, 1 to 2 km below the tropopause.

TABLE 1. Basic and transitional layers of the earth's atmosphere

Basic layers (spheres)	Mean heights of the upper and lower boundaries, km	Transitional layers
Troposphere	0–11	Tropopause
Stratosphere	11–35	Stratopause
Mesosphere	35–80	Mesopause
Thermosphere	80–800	Thermopause
Exosphere (the lower boundary)	800 (over the poles) 1000 (over the equator)	
Theoretical boundary of the atmosphere (according to Smolukhovskii)	28,000 (over the poles) 42,000 (over the equator)	

The stratosphere is characterized by no temperature variation with height.

The mesosphere is generally characterized by a considerable rise in temperature with height in its lower part and fall of temperature in its upper part. The composition of the main gases in the mesosphere is approximately the same as in the lower layers of the atmosphere, but the pressure is very low, varying from 2.5 mm Hg at the lower boundary to 0.01 mm Hg at the upper boundary.

Up to a height of 50 km there is an average rise of temperature of 3 to 4°C per km of height. At a height of 50 to 55 km the temperature is about 0°. The upper part of the mesosphere is characterized by a temperature decrease with increasing height. At a height of 80 km, the temperature is about −70°. Noctilucent clouds are sometimes observed in summer in the upper part of the mesosphere, at heights of 82 to 85 km, indicating the presence of air currents in the mesosphere.

The thermosphere is distinguished by a continuous rise of temperature with height. It differs from the lower layers in other ways as well. The oxygen in the thermosphere is in the atomic state. The disintegration of oxygen molecules into atoms begins approximately at a height of 100 km due to the ultraviolet radiation of the sun. Theoretical calculations show that at heights above 110 to 120 km, the uncombined oxygen contained in the atmosphere can be only atomic. Other gases in the thermosphere include nitrogen, helium, and hydrogen.

One of the phenomena observed in the thermosphere is the aurora polaris, a characteristic glow of the upper layers of the atmosphere due to the corpuscular radiation of the sun. Under the action of the magnetic field of the earth, the flow of corpuscles is deflected toward the poles and, therefore, the aurora polaris is usually observed at high latitudes. The aurora polaris is formed at heights of 100 to 400 km, and sometimes reaches heights of 1000 to 1200 km. A study of aurorae polares may determine most of the physical characteristics of the upper atmosphere.

A characteristic feature of the thermosphere is the presence of a huge amount of charged particles, ions. These ions cause the high electrical conductivity of those sections of the thermosphere in which they occur in their highest concentrations.

The upper layers of the atmosphere, having increased electrical conductivity, are called the ionosphere. The ionosphere is divided into four layers: D, E, F_1, and F_2. The first layer is at a height of 40 to 80 km. It is characterized by the fact that it absorbs radio waves more intensely than it reflects them. This is due to the sufficiently high air density at these heights. The E, F_1, and F_2 layers are observed, on the average, at heights of 100, 200, and 320 km, respectively.

The ionospheric layers are not continuous, but consist of individual ionized clouds. The origin of the ionosphere is mainly due to the corpuscular and ultraviolet radiations of the sun. The layers differ in the concentration of the charged particles and in some other characteristics.

The ionospheric layers reflect some radio waves completely, and pass some others. The latter radio waves can be used for communication with spaceships. For example, reliable communication was established with the first Soviet rocket to the moon on the frequencies 19.997, 19.995, 19.993, and 183.6 Mc, and with the interplanetary station "Mars I", on the frequencies 922.76 and 183.6 Mc.

The exosphere is the uppermost layer of the atmosphere. It is characterized by a temperature increase with height and a very low pressure. Therefore, favorable conditions are created at its upper boundary for the escape of gases from the atmosphere. This occurs when the high temperature causes the thermal velocities of individual molecules to reach and exceed the escape velocity for the given height. As a result of this, such molecules leave the earth's atmosphere and go into interplanetary space. The highest rate of the dissipation is observed for light gases and the lowest rate — for heavy gases. Thus, for example, at a temperature slightly over 700°, the time for complete dissipation of hydrogen is 4 years, and for helium, $1.4 \cdot 10^6$ years.

Only recently a systematic study of the exosphere has begun, using artificial earth satellites and space rockets. The first of these attempts has already led to very interesting and important discoveries. Thus, the experience of the first three Soviet lunar rockets established that even at large distances, the earth is surrounded by a highly rarefied atmosphere, consisting of an ionized gas. This part was called by Soviet scientists the geocorona. The concentration of ions 300 km from the earth's surface is 1 to 2 million per cubic centimeter. In the geocorona it is only a few hundred. For comparison, we recall that at the earth's surface the number of molecules per cubic centimeter is expressed by a twenty-decimal number.

The geocorona is observed on the average up to a distance of 22,000 km from the earth's surface. Its height depends on a number of conditions, mainly the solar activity.

No ionized gas has been detected in interplanetary space at distances over 22,000 km. It is assumed that if there is ionized gas in interplanetary space, its concentration would be considerably lower than several tens of ions per cubic centimeter.

The geocorona is formed by hydrogen atoms constantly escaping the earth's atmosphere. The escaping hydrogen is supplemented by the evaporation of water from the oceans, seas, and rivers. According to calculations, the level of the World Ocean has fallen by several meters during the geological history of the earth due to such evaporation.

Thus, the boundary of space can be considered to be 22,000 km from the surface of the earth. However, spacecraft can make prolonged flights at lower heights, down to 140 to 150 km, where an earth satellite can exist for approximately 1 to 2 orbits. Apparently this height should be considered as the lower altitude limit for space flights.

Investigations of the earth's atmosphere and of space by means of artificial satellites and rockets led to the discovery of the radiation belts of the earth. These are extensive zones of charged particles whose main source is the sun. The earth is surrounded by a cloud of high-energy charged particles, held by its magnetic field. It extends over the entire earth and may be clearly divided into three belts: internal, external, and a third, or as it is sometimes called, the most external (Figure 19).

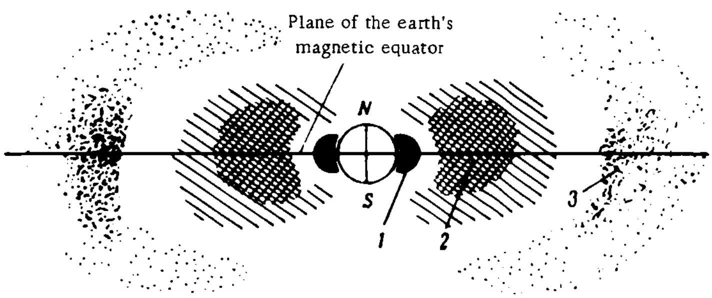

FIGURE 19. Meridional cross section of the radiation belts of the earth: 1- internal; 2- external; 3- most external.

The high intensity of cosmic rays in the upper layers of the atmosphere was almost simultaneously observed by the Soviet scientist S. N. Vernov and the American scientist Van Allen from data of instruments mounted on satellites. Van Allen observed the internal radiation belt, but due to limited data and the fact that the first American satellites did not reach latitudes higher than 35°, very little information on this interesting phenomenon was obtained. The results of the measurements from the third Soviet satellite made it possible to determine the boundaries of the internal radiation belt, to estimate the energy of its particles, and to observe in it longitudinal asymmetry. These resulted in conclusions on the reason for its existence.

The discovery and investigation of the external radiation belt was made by Soviet scientists by means of the third Soviet satellite and space rockets. The report of the Soviet scientists S. N. Vernov, A. E. Chudakov, V. I. Krasovskii, and others, "Investigation of the Cosmic Radiation and Magnetic Field of the Earth and Moon," was awarded the Lenin Prize in 1960. The discovery of the most external radiation belt was also made by Soviet scientists.

What are the radiation belts? The equatorial diameter of the external belt is about 100,000 km. Inside this belt there is a cavity with a diameter of about 40,000 km. The belt is symmetric with respect to the magnetic axis of the earth, and its meridional cross section (see Figure 19) has the form of a sickle whose ends reach the upper layers of the atmosphere at geomagnetic latitudes of 50 to 60°.

The internal belt is situated within the cavity of the external belt. It extends latitudinally approximately 40° on both sides of the magnetic equator. Its height is from some hundreds to 5 or 6 thousand kilometers over the earth's surface /2, 13/.

The belts differ by the nature of their constituent particles: the external belt has electrons with an energy of tens and hundreds of thousands of electron volts, whereas the internal belt has protons with an energy of tens and even hundreds of millions of electron volts.

The third belt of charged particles with relatively low energies is situated approximately 40 to 80 thousand km from the earth. Electrons with an energy from 200 to 20,000 ev have been detected in this belt. The particle current density reaches a maximum in the center of this belt.

It is assumed that the most external belt originated from the penetration of solar particles into the peripheral regions of the earth's magnetic field.

According to a modern hypothesis, the particles of the radiation belts are trapped in the earth's magnetic field and move along the magnetic force lines.

Recent investigations detected a region of increased radiation in the southern part of the Atlantic Ocean, which is related to the magnetic anomaly in this region. The lower boundary of the internal radiation belt descends there to 250 to 300 km above the earth's surface.

According to the opinion of some scientists, the presence of radiation belts is one reason for the nonuniform diurnal rotation of the earth.

It is possible to create artificial radiation belts of the earth by exploding nuclear and thermonuclear devices in space. The nuclear devices exploded by the USA on 20 and 30 August and on 6 September, 1958, in the southern part of the Atlantic Ocean at a height of 480 km led to the appearance of an artificial radiation belt between the internal and external belts. The equatorial diameter of this radiation belt is about 16,000 km. However, such "scientific" experiments, carried out by the USA, hardly contribute to the progress of astronautics.

According to the American press, a thermonuclear device, exploded in the Pacific Ocean over Johnston Island in the summer of 1962, caused more damage in the space near earth than the authorities admitted. The artificially produced radiation turned out to be so strong that it damaged the solar batteries of three American satelites and discontinued radio communication with them. The Atomic Energy Commission and the Department of Defense concluded that the artificial radiation belt formed as the result of this explosion is more intense than was previously assumed, and that it may exist for many years.

The discovery of the radiation belts of the earth suggests that radiation belts are also possible on other planets possessing a magnetic field. This is of great importance in astronautics. The paths of spaceships should normally pass outside zones with intensive cosmic radiation. The existence of radiationless zones at the poles enables spaceships to leave the earth and return to it from interplanetary space.

One may get an idea of the radiational danger in a space flight near the surface of the earth under the lower radiation belt from the magnitude of the total radiation dose received by our first astronauts during their flights. These doses are given by the following figures /42, 43/ (in millirads*):

Yu. A. Gagarin	1
G. S. Titov	10
A. G. Nikolaev	43
P. R. Popovich	32
V. F. Bykovskii	50
V. V. Tereshkova	30

These doses are very small. In fact, the first indications of radiation sickness which do not affect work capacity appear in man when the radiation dose reaches 45 to 90 rad. Severe radiation sickness begins at a dose of 90 to 180 rad /4/. A man normally receives a radiation dose of approximately 0.3 rad in chest irradiation. If the irradiation is received from mobile X-ray equipment, the radiation dose is 2 to 9 rad. The annual radiation dose received by a man from the luminous dial of a wristwatch (at a distance of 0.3 m) is about 0.00004 rad. From the instrument panel in an airplane cabin, containing up to a hundred luminous instruments, the pilot receives an annual dose of up to 0.0013 rad /35/.

§5. Basic Physical Characteristics of the Moon and the Peculiarities of its Motion

The moon, earth's natural satellite, is the large celestial body nearest to us, and is denoted by the symbol ☾.

The mean distance from the earth to the moon is 384,400 km, which is 60.27 radii of the earth. The diameter of the moon is 3473.4 km (0.27 of the diameter of the earth), its mass, $7.35 \cdot 10^{25}$ g (0.01 of the mass of the earth). The volume of the moon is only 0.02 of the volume of the earth, and its density is 3.34 g/cm^3 (0.606 of the density of the earth). Therefore, the gravitational acceleration, and hence the weight of every object on the moon is one sixth of that on the earth. The zero-height circular velocity for the moon is 1.68 km/sec, and the escape velocity is 2.36 km/sec.

The eccentricity of the lunar orbit is small, averaging 0.0549. The lunar orbit can therefore be considered practically circular. The inclination of the lunar orbit is, on the average, 5°09' and the mean velocity of the moon in its orbit is 1.02 km/sec.

The period of one revolution of the moon around the earth is called the lunar month. The time interval during which the moon reaches its original position with respect to the stars after one revolution is called stellar or sidereal month. The time interval during which the moon reaches its original position with respect to the sun, is called synodic month. The motions of the sun and moon as seen from the earth are in the same direction, but since the moon moves faster than the sun, the synodic month is longer than the sidereal month. The first is equal on the average to 29 days 12 hrs 44 min and 2.78 sec, the second, 27 days 7 hrs 43 min and 11.5 sec.

* A rad is a radiation unit corresponding to the absorption of 100 ergs in 1 g of tissue. A millirad is one thousandth of a rad. One rad is equal to 1.12 r (roentgen).

The attraction of the sun, the nonsphericity of the earth, and, to a much lesser extent, the attraction of the planets, give rise to complicated perturbations, or inequalities, in the motion of the moon. The perturbations are manifested in the continuous variation of the elements of the lunar orbit. At the end of the past century, Brown, an American scientist, counted 751 inequalities. Of these numerous inequalities we shall mention only the principal four.

1. Regression (backward motion) of the line of nodes. The line of nodes of the lunar orbit rotates in the plane of the ecliptic in a direction opposite to the motion of the moon along its orbit. The nodes move in the ecliptic by 19.3° per year, as a result of which, in 18.6 years they make a complete rotation with respect to the pole of the ecliptic. Thus the moon has a new path among the stars each month.

2. Direct motion of the line of apsides. The ellipse of the lunar orbit rotates in its plane so that the line of the apsides turns in the same direction as the moon. This rotation has a rate of 40.7° per year, making a complete rotation in 8.85 years.

3. Periodic oscillations of the inclination. The inclination of the plane of the lunar orbit to the ecliptic varies from 4°59' to 5°17' with a period of 18.6 years.

4. Periodic oscillations of the eccentricity. The eccentricity varies from 0.0435 to 0.0715 with a period of 8.85 years. This is also the period of the variation of the semimajor axis of the lunar orbit from 356,400 to 406,730 km.

The inequalities in the motion of the moon lead to highly complicated formulas for the calculation of its coordinates on the celestial sphere. Thus, according to Brown's theory, the longitude of the moon in the ecliptic coordinate system is given by an equation containing 665 terms, and the latitude is given by an equation containing more than 300 terms.

The complicated character of the lunar motion suggests the difficulties which scientists encounter when choosing and calculating flight trajectories for automatic stations and lunar spacecraft to and about the moon.

It is believed that the moon once had a dense atmosphere, but that since the escape velocity for the moon is low, the thermal velocity of individual molecules of the atmosphere became equal to or exceeded it and such molecules left the moon forever. The lower the escape velocity, the faster a heavenly body loses its atmosphere, or, as is said, the faster the dissipation. For this reason, the moon is practically devoid of an atmosphere, even though the ages of the earth and of the moon are the same in the opinion of scientists.

According to the latest data, the density of the atmosphere at the surface of the moon is $2 \cdot 10^{-13}$ of the density of the lower layers of the earth's atmosphere. There is no water on the moon.

The extreme rarefaction of the lunar atmosphere deprives the moon of the protecting shell in which meteorites burn out and are pulverized, and therefore the lunar surface is continually bombarded by meteorites.

Since the moon has practically no atmosphere it is not possible to use the braking property of the atmosphere to reduce the velocity of a spaceship for landing on the moon. To accomplish this, quite powerful braking engines will apparently be required.

From the data of space rockets, Soviet scientists have established that the moon does not have any noticeable magnetic field, and consequently, no radiation belts.

The surface of the moon has been quite well studied. However, the period of rotation about its axis is equal to the period of one complete revolution around the earth (i.e., is equal to a sidereal month), and therefore only half of the lunar sphere is ever seen from the earth.*

The science studying the surface of the moon is called selenography (from selena, the ancient name of the moon). Detailed maps of the visible part of the moon exist on which over 200,000 details of the lunar surface have been plotted. Of these, more than 32,000 details have been named. All formations having a diameter of more than 50 m are now drawn on the map of the visible side of the moon.

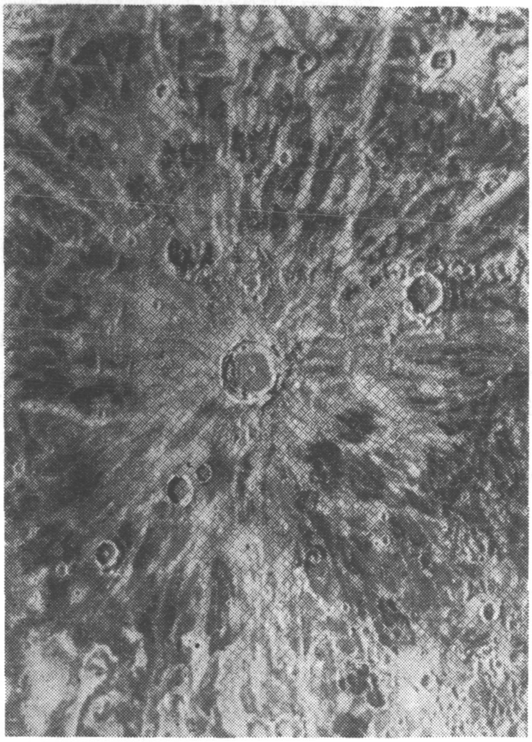

FIGURE 20. Photograph of part of the lunar surface from the crater Copernicus (in the center)

A characteristic feature of the lunar surface is the annular mountains or craters (Figure 20). The largest crater Grimaldi has a diameter of 235 km, and the smallest distinguishable in modern telescopes is 100 to 200 m in

* Actually, about 60% of the entire surface of the moon is seen from the earth due to the phenomenon of libration, a slow tilting of the moon with respect to some average position as a result of the ellipticity of its orbit, the slight inclination of its rotational axis to the plane of the orbit (1°32'), and some nonuniformity of the rotation about its axis.

diameter. The height of the annular banks of the craters reaches 300 to 7000 m. Sometimes a sharp-edged mountain, called a central peak, rises in the center of the crater. Even the unaided eye sees large dark spots of circular form on the surface of the moon, which are the wide plains called "seas." In addition, there are mountain ridges whose height reaches 7 to 8 km; one mountain top is 9 km high.

A system of bright rays, diverging in all directions from some craters (Tycho, Kepler, and others), is seen on the lunar surface. These rays are several kilometers wide and intersect mountains and seas, sometimes extending to distances of up to 5000 km. Furrows and black fissures, probably quite deep, are seen in many places on the lunar surface. A fissure of average dimensions extends 100 km, and is several hundred meters wide and deep.

The reasons for these characteristic features of the lunar topography are not clear yet, but two basic hypotheses exist on the formation of the annular mountains. According to the first hypothesis, the annular mountains are the result of grandiose eruptions of gases which once escaped from the interior of the moon. According to the second, their origin is due to the bombardment of the lunar surface by large meteorites.

Quite recently, the amateur astronomer M. M. Shemyakin noticed an interesting regularity in the positions of the lunar craters. Some crater chains lie along a line close to an arc of a circle, and the area of each crater is half the area of the previous one. The distance between them was also found to obey a certain mathematical law. It is hoped that this discovery will aid scientists to understand the secret of the origin of the lunar craters.

In recent years, astronomers have often observed various changes occurring on the surface of the moon. Thus, for example, in the last few years the area of the crater Linné decreased by a factor of two. The crater Algazen disappeared, leaving no traces; small craters formed on the bottom of the crater Plato, and new, hitherto unobserved fissures have been observed near the crater Messier. On the bottom of the crater Plato, green-grayish spots of an unknown origin appear at sunrise, and in the crater Eratosthenes these spots even appear to be moving /5/.

Thus, powerful processes take place on the lunar surface and in its interior, but their character and causes are as yet unknown. A detailed investigation by spaceships of the topography of the lunar side visible from the earth will make it possible to obtain new data on this problem.

On 4 October, 1959 the third Soviet cosmic rocket was launched in the direction of the moon. The automatic interplanetary station, especially built for photographing the opposite side of the moon, orbited the moon and, according to a precomputed program, photographed it. The images obtained were transmitted to the earth via a special radioengineering system.

The time chosen for the photographing made it possible to obtain pictures of that part of the lunar surface invisible from the earth and of a small area with known formations. The photographs of a part of the known region of the moon allowed us to compare unobserved features on the opposite side of the moon to those already known and in this way determine their selenographic coordinates.

Analysis of the photographs shows that mountainous regions prevail on the invisible part of the lunar surface, but there are very few seas similar to those of the visible part. The pictures obtained of the opposite side of the moon enabled the first Russian atlas of the lunar surface and a lunar globe. The optical reflections of the moon indicate that the lunar surface is irregular. It is covered all over, including the plains, by numerous irregularities, which cannot be seen from great distances. Until recently, it was assumed that the lunar surface is covered by a dust layer from several centimeters to tens of centimeters thick. However, according to the latest data, the surface of the moon is a very light porous substance. This fact was established as a result of the analysis of radio emissions of the moon and was confirmed by calculations. It was established, for example, that from 1.72 to 86 kg of meteorite dust falls annually on each square kilometer of the earth's surface, and about 0.34 kg should fall annually on each square kilometer of the lunar surface. This amount of dust, uniformly distributed over the lunar surface, produces a 1 cm layer after 100 million years. If we take into account that the age of the rocks forming the surface of the moon is 500 million years, then the dust layer on the surface of the moon should be about 5 cm thick. However, the existence of a dust layer on the moon is now denied by a number of scientists.

Some scientists assume that the moon has a hot core in order to explain the temperature rise discovered in the deep layers of the moon. Due to the absence of an atmosphere, three times the amount of solar radiation falls on a unit area of the lunar surface as on a unit area of the earth's surface.

During the two-week lunar day the surface of the moon, illuminated by the solar rays, reaches a comparatively high temperature, and during the long lunar night becomes extremely cold, due to the absence of a protecting layer such as the atmosphere which preserves the heat on the earth during the night. Thus, it can be affirmed in advance that the temperature of the lunar surface undergoes extremely sharp variations. To date the temperature in various places of the lunar surface has been measured quite accurately.

Measurements carried out in one of the observatories made it possible to obtain the following figures. The temperature in the central parts of the full moon disk, i.e., at a point in direct sunlight, was found to be 134°C. In the middle, between the center and edge of the lunar disk, it is 122°C, at a distance of 0.75 r [r = radius of the moon] from the center of the lunar disk, 102°C, at a distance of 0.9 r, 77°C, and at the very edge of the disk, 67°C. Theoretical values closely approximate these. The temperature of the night side of the moon is about −150°C.

The second Soviet cosmic rocket sent to the moon, on which nine counters to measure the radiation level near the moon were mounted, established that the moon does not have radiation belts similar to the earth.

These are in general the unfavorable conditions which man will encounter on the moon.

Much about the moon is now definitely known, but the rest is, for the time being, pure guesswork. No doubt that the further development of science and engineering as a whole, as well as the further progress of astronautics, will considerably add to our knowledge of the moon.

The moon is the celestial body closest to the earth, and therefore it is obvious that in the near future, manned spaceships will be directed to it for a detailed study.

§6. The Inferior Planets

The planets of the solar system whose orbits lie within the orbit of the earth are called the inferior planets, and those outside, the superior planets. There are two inferior planets, Mercury and Venus.

The conventional designation of the planet Mercury is ☿.

The equatorial diameter of Mercury is 4720 km, 0.37 of the earth's diameter. Its volume is 0.050, and its mass 0.037 of the mass of the earth. Its average density is 4.1 g/cm^3 (0.74 of the average density of the earth). The gravitational acceleration on the surface of Mercury is 2.55 m/sec^2, which is 0.26 of the gravitational acceleration on the surface of the earth. The zero-height circular velocity for the surface of Mercury is about 2.7 km/sec, and the escape velocity at the surface is 3.8 km/sec.

The distance of Mercury from the sun varies widely from 46 to 70 million km, the mean being 57.9 million km. The eccentricity of the orbit is almost 0.206. The distance from the earth to Mercury varies from 82 to 217 million km. Its relatively small distance from the sun is the reason for the very high temperature on the illuminated part of its surface. The temperature at a point in direct sunlight is +412°C. This practically corresponds to the melting point of zinc.

The mean linear velocity of Mercury in its orbit is 47.83 km/sec, the sidereal revolution period is 87.97 days, and the mean synodic period is 115.88 days. The inclination of the orbit is 7°00'14", and the period of rotation about its axis is 87 days 23 hrs 16 min. The period of Mercury's rotation about its axis is equal to the period of its revolution around the sun, and consequently Mercury always keeps the same side facing the sun, as the moon does to the earth. The inclination of Mercury's equator to the plane of its orbit is unknown.

Similar to the moon, Mercury has phases. The study of its spectrum leads one to assume that the atmosphere of Mercury is extremely rarefied. Mercury does not have any satellites.

The second inferior planet is Venus. Its conventional designation is ♀.

Venus resembles the earth in many respects. Its equatorial diameter is 12,374 km, which amounts to 0.97 of the equatorial diameter of the earth. Its mass is 0.826 of the mass of the earth, its volume 0.90 of the volume of the earth, and its average density 5.1 g/cm^3 (0.92 of the average density of the earth).

The gravitational acceleration on the surface of Venus is 8.83 m/sec^2, which is 0.90 of the earth's gravitational acceleration. A man who weighs 70 kg on the earth will weigh slightly less than 63 kg on Venus. The zero-height circular velocity for Venus' surface is 7.2 km/sec, and the escape velocity at the surface is 10.2 km/sec.

The mean distance of Venus from the sun is 108.1 million km; its orbit has the smallest eccentricity of all the planets of the solar system, 0.007. The minimum distance from the earth to Venus is only 40 million km. Similar to the moon and Mercury, Venus has phases. Venus is nearer to the sun than the earth, and therefore receives from it slightly over twice as much heat and light as does the earth. Venus does not have any satellites.

The mean linear velocity of Venus in its orbit is 34.99 km/sec, and the angular velocity, 1°36' per day. The sidereal period of revolution is 224.7 days and the mean synodical period is 583.92 days. The inclination of the orbit is 3°23'39", and the inclination of the equator to the plane of the orbit is not known accurately.

A temperature of −23°C has been measured on the dark part of Venus /27/. According to radar measurements in the USSR and the measurements of the American space probe "Mariner II", the temperature of the external part of the Venusian atmosphere is about 600°C. This proves that Venus cannot always turn the same side to the sun, for its night hemisphere would then be colder.

In spite of its relative closeness to the earth, Venus hides on it many riddles. To this day it is still unknown what the rotational velocity of Venus is and what is the length of its day. The reason for this doubt is that the surface of the planet is covered by a thick layer of clouds and a direct determination of its rotational period by topographical details is therefore impossible. Indirect methods of determining the rotational period lead to highly contradicting results. According to the results of the radar measurements by the Soviet scientists, made in 1961, the rotational period of Venus was close to 11 terrestrial days. In 1963 the Soviet scientists A. D. Kuz'min and A. E. Salomonovich presented the astronomical journal of the Academy of Sciences of the USSR (Vol. XL, issue 1) with the following data, "The true (stellar) rotational period of Venus is 69 terrestrial days and the solar period is 98 days. These data were obtained from analysis of the radio emission of Venus in the millimetric and centimetric wave bands."

Venus has a dense atmosphere whose existence was for the first time proved by M. V. Lomonosov from analysis of the rare phenomenon observed by him on 26 May, 1761 — the passage of Venus over the solar disk. Until recently, neither oxygen nor water vapor had been observed in the atmosphere of Venus. Only considerable amounts of carbon dioxide and traces of nitrogen and its compounds with oxygen were seen. The absence of free oxygen in the Venusian atmosphere indicates the absence of a thick vegetation. It is assumed that the conditions on Venus are close to those which existed on the earth before life appeared on it.

Recently a report appeared in the Soviet press that molecular oxygen has been detected in the upper layers of the Venusian atmosphere. This discovery was made by V. K. Prokof'ev of the Crimean Astrophysical Observatory on the basis of unique spectra of Venus, obtained on a large solar telescope and a special spectrograph.

It is thus now established that there is carbon dioxide, oxygen, and nitrogen in the Venusian atmosphere. The presence of nitrogen was discovered by analysis of the spectrum of the night-sky glow of Venus, obtained by the Soviet astronomer N. A. Kozyrev. This fact compels us to approach the problem of life on Venus in a new way; it is now assumed that organic life exists on this planet.

By means of modern telescopes it is possible to distinguish details approximately 12 km wide on the surface of Venus, but in view of the dense cloud cover, we do not know the topography of Venus. However, some scientists believe that the surface of this planet is covered by a continuous ocean.

In view of its relative closeness, Venus is a very tempting object for investigation by spacecraft.

§7. The Superior Planets

The superior planets of the solar system in the order of their distance from the sun are: Mars, Jupiter, Saturn, Uranus, Neptune, Pluto. The conventional notation of the superior planets is: Mars— ♂, Jupiter— ♃, Saturn — ♄, Uranus — ♅, Neptune — ♆ and Pluto — ♇ (PL). The physical characteristics of these planets and the elements of their orbits are given in Tables 2 and 3.

TABLE 2. Physical characteristics of the superior planets of the solar system

Name of planet	Equatorial diameter		Flattening	Volume ♁=1	Mass ♁=1	Density		Gravitational acceleration on the surface,* m/sec²	Cosmic velocities for the surface, km/sec		Temperature at a point under the sun, °C	Inclination of the equator to the orbit
	♁ = 1	in km				♁=1	in g/cm³		zero circular velocity	escape velocity		
Mars	0.54	6,889	1 : 105	0.157	0.108	0.69	3.8	3.73	3.54	5.0	+ 30°	25° 10'
Jupiter	11.14	142,113	1 : 16	1,295	314.8	0.25	1.38	25.9	43.5	61.0	– 140°	3° 01'
Saturn	9.4	119,915	1 : 11	745	95.2	0.13	0.72	11.09	26.0	36.7	– 150°	26° 45'
Uranus	4.0	51,028	1 : 19	63	14.6	0.23	1.3	8.24	15.3	21.6	– 180°	98° 00'
Neptune	4.3	54,885	1 : 40	78	17.3	0.22	1.2	11.18	17.0	23.8	– 210°	29° 36'
Pluto	0.46	5,870	?	0.098	< 1	?	?	?	?	?	–220°	?

* Disregarding the centrifugal force. On Jupiter's equator the gravitational force is reduced by 9%, on Saturn's equator, by 16%.

Of the superior planets the most studied is Mars. As the eccentricity of its orbit is relatively large, the distance from Mars to the sun varies within wide limits — from 206.6 million km to 249 million km. The distance from the earth to Mars varies from 55.5 million km to 400 million km. Mars is nearest to the earth at the time of oppositions, which occurs approximately every two years. Every 15 or 17 years Mars is in so-called great opposition, when opposition and perihelion nearly coincide. For this reason it comes particularly close to the earth. The last great opposition of Mars was in 1956; the next will be in 1971.

The Martian days, as seen from Table 3, are only slightly longer than the terrestrial ones. The inclination of Mars' axis to the plane of its orbit is almost the same as that of the earth (about 65°), and therefore the annual change of seasons on Mars and the illumination conditions in different seasons are approximately the same as on the earth, but the Martian year is almost twice as long as the terrestrial one, and due to its remoteness from the sun, Mars receives only from 36 to 52% of the light and heat which the earth receives.

The science studying the surface of Mars is called areography.* Many stable details are observed on its surface which make it possible to draw a

* From the Greek name of Mars — Ares.

TABLE 3. Orbital elements of the superior planets of the solar system

Name of planet	Average distance from the sun		Orbital velocity, km/sec	Sidereal revolution period			Mean synodic period in days	Period of rotation about the axis	Eccen-tricity	Inclination of orbit
	in a. u.	in million km		in tropical years*	in years and mean days					
Mars	1.52369	277.8	24.11	1.88089	1	231.730	779.94	24 hrs 37 min 23.6 sec	0.09336	1°51'00"
Jupiter	5.20280	777.8	13.05	11.86223	11	314.84	398.88	9 hrs 55 min 41 sec **	0.04842	1°18'21"
Saturn	9.53884	1,426.1	9.64	29.45772	29	166.98	378.09	10 hrs 14 min 24 sec	0.05572	2°29'25"
Uranus	19.19089	2,869.1	6.80	84.01529	84	007.45	369.56	10 hrs 42 min	0.04718	0°46'23"
Neptune	30.07067	4,495.7	5.43	164.78829	164	280.30	367.49	15 hrs 48 min	0.00857	1°46'28"
Pluto	39.45774	5,899.1	4.73	247.6968	247	255.1	366.74		0.24864	17°08'34"

* Tropical year is the term applied to the time interval between two successive transitions of the center of the sun through the vernal equinox. It is equal to 365.24220 mean days.

** For the equatorial zone — 9 hrs 50 min 30 sec.

quite detailed aerographical map. The smallest of the details on the Martian surface observable from the earth with a modern telescope is about 16 km wide.

Investigations have shown that there are no large mountains on Mars, its surface is very smooth and even. If there are elevations, they are not mountains, but rather plateaus with a height of no more than 1000 m. Four types of large surface details are observed on the disk of Mars: polar caps, "seas," "continents," and "canals" (Figure 21).

The polar caps are white spots on the poles of the planet. These formations are highly variable; they decrease with the arrival of the Martian summer and increase in winter. The polar caps are possibly a thin ice crust only several centimeters thick /27/.

If the polar ice of Mars were melted, all the resulting water would constitute a lake no greater than Lake Lagoda.* Water has also been observed in the Martian atmosphere. If there are open water basins in the equatorial regions of the Martian surface, then, according to some calculations, their diameter does not exceed 300 m.

The Martian "seas" are darker than the whole surface of Mars, their dimensions and color depend on the Martian season.

The "continents" are huge formations on the Martian surface, reddish-orange in color, extending over approximately two thirds of its surface. Most probably, these are deserts with a soil stained by ferric oxides.

The Martian "canals" are regular thin lines intersecting the reddish surface of the "continents" and "seas." The Martian canals were discovered by the Italian astronomer Sciaparelli in 1877 during one of the great oppositions of Mars. He is the one who proposed the idea that the network of the Martian canals, striking by its rectilinearity, is a gigantic irrigation system, constructed by intelligent inhabitants. Later, at the intersection places of the canals enigmatic dark circular spots, oases, were discovered. As many as seventeen canals converge into some of the oases. Seasonal variations of the canals have also been observed. With the arrival of the Martian winter, the canals become faded and many of them completely disappear. At the beginning of the spring they reappear. First the canals which are situated close to the thawing polar cap become noticeable followed by the more remote ones. The darkening of the canals goes from the pole to the equator. In the equatorial belt of Mars, where one darkening is replaced by another, the canals are always visible.

In 1924 scientists succeeded in photographing the canals of Mars and making the first photomap. In 1939 hundreds of photographs of Mars were obtained on which about 500 canals were observed. The total number of canals discovered exceeds 1000. The photographs also showed seasonal variations of the canals.

Some scientists assume that there is vegetation on Mars. The opponents of this assumption refute this possibility with a series of quite convincing arguments. For example, the green pigment of terrestrial plants very strongly scatters infrared rays, and those places on Mars which were considered a plant cover do not display this property. The spectrum of the terrestrial green is characterized by a wide dark band in the extreme red rays, the absorption band of chlorophyll, which is not observed in the spectrum of Mars.

* [Lake Lagoda has an area of about 7000 sq. m.]

The Soviet scientist G. A. Tikhov, studying terrestrial plants in severe high-mountain conditions, i. e., in temperature conditions approaching Martian conditions, succeeded in proving that by prolonged adaptation, the vegetation of Mars could acquire properties other than those known of the terrestrial flora. The study of the Martian vegetation marked the birth of a new science — astrobotanics.

An atmosphere is observed on Mars, but the atmospheric pressure at the surface is small, from 100 to 200 mm. This pressure is observed on the earth at a height of 15 to 20 km. The composition of the Martian atmosphere has not been accurately established. However, much nitrogen and some carbon dioxide have been observed. As a result of observations of Mars and the study of spectrograms, Professor N. A. Kozyrev succeeded in proving the existence of water in the form of ice and snow.

The lower gravitational acceleration on the surface of Mars as compared with the earth leads to an interesting phenomenon. The pressure and density of the atmosphere fall off more slowly with increasing height than on the earth. At a height of approximately 25 km, the pressure of the Martian atmosphere is equal to the pressure of the terrestrial atmosphere at this level. At large heights the pressure and density of the Martian atmosphere exceed those of the terrestrial atmosphere. Consequently, spaceships flying to Mars will experience a stronger deceleration in the upper layers of the Martian atmosphere than in the upper layers of the terrestrial atmosphere, and therefore the descent to the surface of Mars will differ from the descent to the earth.

From photographs of Mars made during the opposition of 1939, Professor N. N. Sytinskaya obtained the following data on the Martian atmosphere. The total mass of gas per unit area of the Martian surface is .22 times that on earth. A mercury barometer on the surface of Mars would show a pressure of 170 mm of Hg, and an aneroid barometer (whose readings depend not only on the amount of gas, but also on the gravitational force), only 65 mm of Hg. At this pressure, water boils at 43°C.

The climate of Mars is more severe than that of the earth. This is due to extreme rarefaction of the atmosphere and the greater distance of Mars from the sun. On the equator of Mars at about midday, the temperature reaches 5 to 20°C. At night the temperature falls below 0° and at dawn reaches −46°C. In winter, near the southern pole, the temperature is thought to be −60°C.

Mars has two satellites, discovered in 1877 by the astronomer Hall. The nearest satellite, Phobos, has a diameter of only 16 km. It completes a revolution around the planet in 7 hrs 39 min 14 sec, i. e., its rotation is considerably faster than the rotation of Mars about its axis, and therefore Phobos rises above the horizon of a Martian observer on the west, moves against the diurnal motion of the stars and sets in the east. Such a motion of a planet's satellite is at present the only known case in the solar system. The mean distance of Phobos from the surface of Mars is 9376 km. The second satellite, Deimos, has an even smaller diameter, only 8 km. The distance from the center of the planet to Deimos is 23,500 km.

After analyzing the motion of the Martian satellites, the Soviet scientist I. S. Shklovskii suggested that they are of artificial origin.

It is clear that, like the moon and Venus, Mars will soon be an object for study of automatic interplanetary stations, and later of spaceships.

The beginning of this study was set by the launching of the Soviet automatic interplanetary station "Mars I" on 1 November, 1962.

The planets Jupiter, Saturn, Uranus, and Neptune have huge dimensions as compared with the earth. They are called giant planets. The most massive of them is Jupiter. Its mass is 318 times, and its volume 1300 times the mass and volume of the earth, but the average density is almost one-fourth the density of the earth. Varying belts, parallel to the equator, which were later recognized to be clouds, have been observed on the surface of Jupiter.

By spectral analysis, it has been established that the atmospheres of all the four giant planets consists of ammonia and methane (marsh gas). The temperature of Jupiter's atmosphere is very low (from −100 to −140°C).

Jupiter rotates very rapidly about its axis, not as a solid body, but by "zones." Each zone has its own rotational velocity. Jupiter's axis of rotation is almost perpendicular to the plane of its orbit, and therefore there is essentially no change of seasons on this planet.

Jupiter and the other giant planets are covered by dense clouds, and hence reliable data on the character of their surfaces and their internal structure are not available. It is most probable that a thick layer of ice and frozen gases lies under the cloud cover of these planets, and that the internal part is a solid nucleus. The low average density of the planets is apparently due to their extensive atmosphere. It is also assumed that the giant planets contain much hydrogen.

At present, 12 satellites of Jupiter are known. The names and relative dimensions of the satellites are shown in Figure 22.

On Saturn, cloud belts are less noticeable. Their color is mainly brown. Saturn is the second largest planet. The form of the planet in a telescope, the temperature of its atmosphere, and its rotation indicate that the physical structure of the two largest planets is almost identical. The axis of rotation of Saturn is inclined at an angle of 63° to the plane of its orbit, and therefore Saturn shows a change of seasons.

The peculiarity of Saturn is its ring, being a unique formation in the entire solar system. The ring is situated in the plane of the planet's equator but does not touch the planet. Its dimensions are tremendous: the external radius of the ring is 137.5 thousand km, which is 2.3 radii of the planet; the thickness of the ring is not more than 15 to 20 km.* It is assumed that the ring has a meteorite structure. The sizes of the ring particles vary from tenths of a micron to tens of centimeters.

Through strong telescopes the ring is seen to consist of two parts. The interval between the rings is about 5000 km. The mass of Saturn's ring is estimated to be 1/27,000 of Saturn's mass.

Nine satellites of Saturn are known (see Figure 22). The satellites are situated at distances from 185,000 km to 12.9 million km from the center of the planet.

On the greenish disk of Uranus, it is almost impossible to distinguish any details. The axis of rotation of Uranus is inclined to the plane of its orbit at an angle of only about 3°. This leads to a very strange and unique phenomenon in the solar system. Uranus always revolves around the sun almost "lying on the side." As a result there is a very peculiar change of seasons, and the "day" and "night" may last for several years. Five satellites of Uranus are known.

* According to other data the thickness of Saturn's ring is considerably smaller.

FIGURE 21. Appearance of Mars on a color photograph

It is also impossible to distinguish any details or spots on the disks of Neptune and Pluto, discovered in 1930, and we still know very little about these planets.

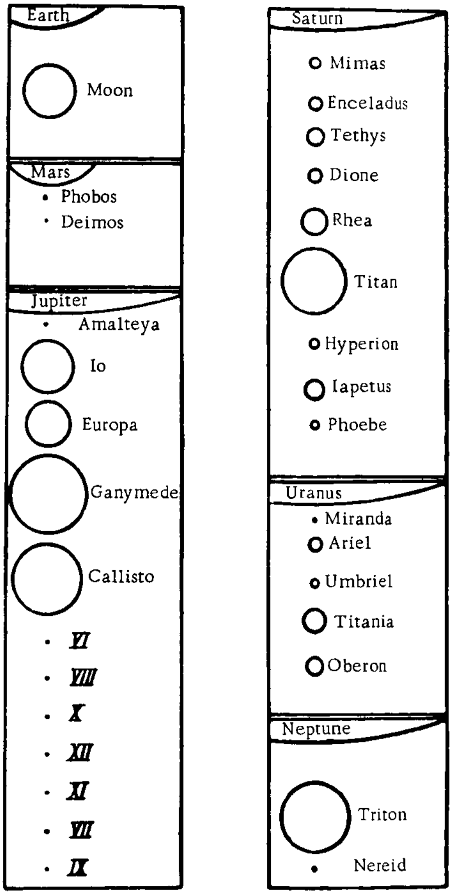

FIGURE 22. Relative dimensions of the satellites of the planets

Flight to the superior giant planets is a remote prospect. Its accomplishment requires, first of all, the development of engines capable of giving the spaceship considerably higher velocities than may be achieved at present. However, this is not the only difficulty preventing flights to the giant planets. Such a flight requires overcoming the belt of minor planets, the asteroids.* The return flight requires tremendous energy expenditures, as the escape velocity for the nearest of the giant planets, Jupiter, is 61 km/sec, about 5 times that for the earth. Trajectory correction and the necessity to transmit scientific information from such spaceships to the earth requires the development of super-long-distance space-communication systems. In addition, it is necessary to develop methods for protecting

* The asteroids are discussed in the next section.

the astronauts from the extremely low temperatures on the surface of these planets and from what is perhaps more complicated, the tremendous gravitational force. The gravity acceleration on the surface of Jupiter is 25.9 m/sec^2. A 70-kg astronaut will weigh on the surface of this planet almost 186 kg!

However fantastic the achievements of science and engineering, however great their possibilities, it is hardly possible to expect that man will visit these planets. Apparently, they will be studied only by means of automatic interplanetary stations and automatic stations with scientific instruments landed on their surface. However, this too is a very interesting and complicated scientific and engineering problem.

These are in short the characteristics of the superior planets of the solar system.

§8. Minor Planets (Asteroids), Comets, Meteor Streams, and Meteors.

The orbits of the minor planets, or asteroids, lie mainly between the orbits of Mars and Jupiter, although some of them, for example Hermes, approach the orbit of Venus at perihelion, and others (Icarus) even intersect the orbit of Mercury (Figure 23). Most of the minor planets are about 2.5 a.u. from the sun.

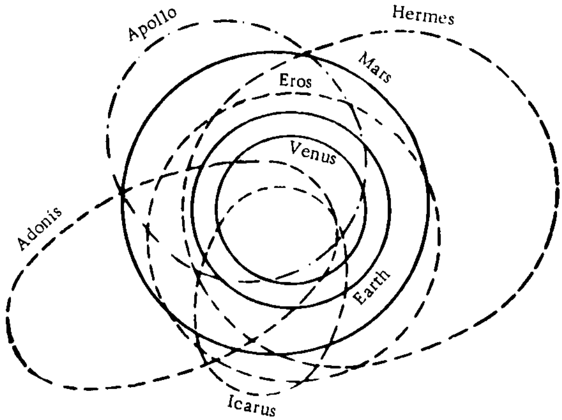

FIGURE 23. Orbits of some minor planets (asteroids)

By 1 January, 1953, 1586 minor planets were known. It is assumed that their total number exceeds 50,000. For the overwhelming majority of the minor planets, the orbit eccentricity is small, but in the case of some the eccentricity is very large (for example 0.83 in the case of Icarus).

The largest of the minor planets, Ceres, is 770 km in diameter has a mass approximately 1/8000 of the mass of the earth, and the smallest of the known minor planets have a diameter of about 1 km and, by their dimensions, approach the large meteorites (Figure 24).

The short-period oscillations of the luminosity of many minor planets indicate that they rotate and that they have an irregular geometric form.

Some minor planets approach closer to the earth than all the other celestial bodies, excluding the moon. Thus, Hermes reaches only 600,000 km from the earth, Apollo, 5,000,000 km, Amur, 15,000,000 km, and Eros, 23,000,000 km. These and some other asteroids may serve in the near future as objects of study by means of automatic interplanetary stations.

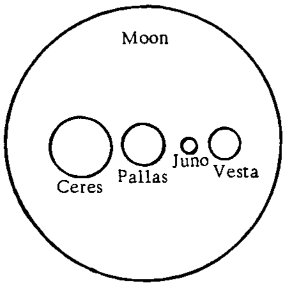

FIGURE 24. Dimensions of some large asteroids as compared with the moon

Comets are the most astonishing members of the solar system. Their geometrical dimensions are very large, sometimes even larger than the sun. Comet tails extend over many hundred million kilometers. (In the case of one of the observed comets, the tail extended over 900,000,000 km.) The mass of comets is small, and therefore their density is negligible. The mass of the most "massive" comet does not exceed 10^{-10} of the mass of the earth, and its nucleus has a diameter equal to several tens of kilometers.

According to the latest data, the densest part of a comet, its nucleus, constitutes a single body of small dimensions. It contains stony and ferrous materials, as well as gases (methane, ammonia, carbon dioxide, cyanogen, and others) in the solid state.

Comet orbits have, as a rule, large eccentricities, as well as large inclinations to the plane of the ecliptic. Comets move both in the direction of the general motion of the planets, and in the inverse direction. Their revolution periods around the sun are also diverse. Forty comets with periods from 3 to 76 years are known, and about 40 comets with periods of about 200 years.

Meteors, or falling stars, are seen as visual phenomena in the earth's atmosphere, flashes caused by small particles, meteoric bodies, entering the atmosphere of the earth at huge velocities. These velocities at the moment of encounter with the earth's atmosphere relative to the moving earth vary from 13 to 70 km/sec. Heated by the friction with the air, the meteoric bodies become incandescent and pulverize in the atmosphere. The glow occurs mainly at heights of 50 to 130 km from the earth's surface.

The earth encounters, in 24 hours, 75 million meteors which are visible at night to the unaided eye. The total number of meteors, including the faintest, apparently amounts to hundreds of billions.

If the meteor is sufficiently large, it produces in the atmosphere a very rare phenomenon called a bolide. A bolide is observed in the form of a large brightly luminous sphere with a long trail. When the meteor has considerable mass and relatively low velocity of encounter with the earth, the mass of the meteor has no time to disintegrate completely in the air. The remaining part of the meteor falling to the surface of the earth is called a meteorite.

It had been thought that not more than 1000 tons of meteoric material falls annually on the surface of the earth. Data obtained from artificial earth satellites made it possible to conclude that the daily "dose" of meteoric material arriving at the surface of the earth is considerably larger, equal to about 30,000,000 tons.

When the earth's orbit intersects the orbit of a shower of meteoric particles, one observes a meteor stream. The particles forming the meteor showers move in elliptical orbits around the sun. The origin of these showers is connected with the disintegration and scattering of comets in space.

Some meteor-shower orbits are known and have been studied (Table 4).

TABLE 4. Some data on the basic constants of meteor streams

Stream	Date of stream maximum	Particle velocity with respect to the earth, km/sec	Hourly number of meteors at the maximum
Quadrantids	3 January	42.4	40
Lyrids	22 April	49.1	10
η - Aquarids	4 May	67.0	36
β - Cassiopeids	27 July	—	—
δ - Aquarids	30 July	43.0	14
α - Capricornids	1 August	25.1	—
Perseids	12 August	60.3	55
Orionids	22 October	67.8	10
Southern Taurids	1 November	30.2	6
Northern Taurids	8 November	31.5	
Leonids	19 November	71.7	8
Geminids	13 December	36.4	60
Ursids	22 December	31.2	10—20

The major planets, mainly Jupiter, exert a disturbing influence on meteors, modifying their orbits. This may cause the earth to miss some of the previously known meteor showers and, on the contrary, to meet with a previously unobserved shower.

In recent years, radar has been used for observing meteors and meteor streams. As a result some new meteor streams have been discovered.

Besides meteor streams, single meteors, called sporadic meteors, are also encountered.

Meteors are a serious hazard for spaceships. Sporadic meteors whose paths cannot be taken into account in choosing and calculating the trajectory of a spaceship are a particular danger. Encounter with meteors may result in destruction of the hermetic sealing of the cabin and damage to the ship's equipment.

According to recent data, the velocities of meteoric particles within the solar system reach 50 to 75 km/sec.

The masses and dimensions of meteoric bodies vary from microdust particles to meter-size chunks of stone or metal. The average density of meteoric particles, according to the opinion of some scientists, is 3 to 3.4 g/cm^3.

The meteor danger in a space flight is estimated by the probability of the spaceship encountering meteors of a given mass and by the piercing power of the meteors. The mean time, in seconds, between two successive encounters of a spaceship with meteors of a mass equal to or larger than M can be determined by the formula /28/

$$t = 2\cdot 10^{18}\,\frac{M}{SV_{\text{av}}},$$

where M is the mass of the meteor, in g; S is the area of the midship cross section, in m^2; V_{av} is the average meteor velocity, equal to 30,000 to 50,000 m/sec.

For example, if we take a comparatively small cross section of the spaceship, 1 m^2, then the probability of the spaceship encountering meteors of various masses will be: with meteors of mass $2.5 \cdot 10^{-9}$ g and more, once in 1.5 days; with meteors of mass $2.5 \cdot 10^{-7}$ g and more, once in 5 months; with meteors of mass $2.5 \cdot 10^{-5}$ g and more, once in 39 years.

The probability of encountering larger meteors is negligibly small. A spaceship with the cross-sectional area of 1 m^2 may encounter a meteor of a diameter of 1 mm once in 2500 years, and a meteor of a diameter of 5 mm, once in 330,000 years. As the cross-sectional area of the spaceship S increases the probability of encounter with meteors increases, and the time interval between two successive encounters with meteors of the same mass decreases, as seen from the formula.

The penetration depth of meteors into the metallic shell of a spaceship is 8 to 20 diameters of the meteor. Appreciable destructions of the shell can be caused by meteors with a mass not less than $2.5 \cdot 10^{-7}$ to $2.5 \cdot 10^{-5}$ g, but, as was already said above, the probability of encountering such meteors is comparatively low /28/.

Thus, we can draw the following general conclusion on the meteor danger to space flight: for spacecraft with a small cross section and short flight duration, the meteor danger is small; for spaceships intended for prolonged flights to the depths of the universe, the meteor danger increases due to the large cross-sectional area and the long flight duration. Thus, if the cross-sectional area of the spaceship will be equal, for example, to 10 m^2, then the above-given flight time for one encounter with meteors of a given or larger mass should be reduced by a factor of 10. But for prolonged space flights, ships with an even larger cross-sectional area will apparently be built. Consequently, it is necessary to find methods for combating the meteor danger, and to develop effective systems of protection.

New interesting data on the distribution of meteoric matter in the space beyond the earth's orbit have been obtained from the automatic interplanetary station "Mars I." At a distance of 6000 to 40,000 km from the earth, "Mars I" intersected the meteor swarm of the Taurids. Its instruments then recorded approximately one impact of a meteoric particle in two minutes. At a distance of 20 to 40 million km from the earth, it intersected another, as yet unknown and unobserved, meteor swarm. There too, approximately the same density of meteor bodies was recorded.

Much work is still necessary to estimate more accurately the meteor danger and to develop effective protection for future astronauts.

Are there other planets in the universe in addition to those which revolve around the sun? Do there exist "earths" about other stars inhabited by intelligent beings? Are the solar system and the earth a lucky exception in the infinite universe? These problems have interested scientists and served as a subject of hot debates, discussions, and from time to time, fantastic assumptions. For a long time only guesses and assumptions were made regarding the existence of other planetary systems. For example, Bruno Giordano [1548 — 1600] proposed the idea of the existence of other stars, and an infinite number of "earths," inhabited by intelligent beings.

Beginning in 1937 a number of astronomers, including the Soviet scientist A. N. Deich, observed small irregularities in the motion of some stars. It was found that these stars move in space along complicated curves as though slightly moving from side to side. This peculiarity in the motion of stars can be due only to the motion of their satellite planets, whose attraction causes the bending of the star trajectories. Calculations showed that the masses of the satellites of these stars are similar to the masses of the major planets of the solar system.

Investigations show that the sun and some other stars have approximately the same stellar age and a similar temperature of their external layers. Thus it may be assumed that planets revolve around them. There are very many such stars, even in the part of the universe we can see, and therefore the number of planetary systems should also be large. By the calculations of some astronomers, for each one million stars, there should be on the average one inhabited planetary system, and this means that in the Galaxy there are about 150,000 planetary systems where life exists.

It may be that one of these "earths" is the invisible planet of a comparatively close star "Cygnus-61," situated only ten light years from us.

The possibility of life on other planetary systems is the greatest riddle of the universe. Obviously much time will pass before mankind will obtain direct proofs of the correctness of the assumptions made.

Chapter II

TRAJECTORIES OF COSMIC SHIPS

§1. Celestial Mechanics — the Science of the Laws of Motion of Celestial Bodies and Spaceships

All celestial bodies are in continuous motion. The planets orbit the sun and are in turn orbited by their satellites. The sun also revolves with respect to the center of the Galaxy.

All these motions are caused by one basic property of matter, the mutual attraction of physical bodies. This property is expressed by the law of universal gravitation. It is natural that the motion of artificial celestial bodies, spaceships flying by inertia, should be governed by the same law.

The study of the motion of celestial bodies due to their mutual attraction is the science of celestial mechanics, knowledge of whose elements is therefore essential to explain the motion of spaceships.

The movement of a body can be represented by three independent motions — rectilinear motion of the center of inertia, rotation about the center of inertia, and deformation of the body. The rectilinear motion of celestial bodies is the principal subject of celestial mechanics.

The "n-body problem" arising in this study can be formulated as follows: determine the laws of motion of n point masses, attracting each other by the law of universal gravitation. The general solution is complicated. Newton found a simple solution for the problem of two bodies, but for three bodies this is impossible, as the unsuccessful efforts of the greatest 18th and 19th century mathematicians indicate.*

But the lack of a complete solution of the n-body problem is not an obstacle to study of the motion of celestial bodies. In almost all practical cases, the attraction of one body far exceeds the attraction of all the remaining bodies. Thus, for example, the dominating factor in the earth's motion is the attraction of the sun, the dominating factor in the motion of the moon and of the earth's satellite spaceships is the attraction of the earth, and so on. Therefore, by taking into account only the attraction of the dominant body, i. e., solving the problem of two bodies, a first approximation to the actual motion of the body is obtained. This first approximation is called unperturbed or Keplerian motion. The study of unperturbed motion is the basic problem of celestial mechanics.

To obtain the true motion, it is necessary to take into account the influence of other bodies which was disregarded in the analysis of unperturbed motion. The influences of other bodies on Keplerian motion are called perturbations, and the corresponding branch of celestial mechanics is called the theory of perturbations.

* Particular three-body problems can be solved comparatively simply. See, for example, /41/.

In connection with the realization of space flights, new and complicated problems arise in celestial mechanics, such as the study of perturbations in the motion of artificial satellites of planets, which move at small distances from the surface of the planets; the transfer of spaceships from one orbit to another; the motion of spaceships whose orbits pass successively near two and more celestial bodies, etc. Until these problems are solved, further development of astronautics is impossible.

The law of universal gravitation was for the first time formulated by Isaac Newton in his work "Mathematical Principles of Natural Philosophy" (1687). His discovery was based on seventeenth-century acheivements of astronomy and mathematics, and in particular, on the laws of planetary motion, obtained between 1609 and 1619 by Kepler.

According to the law of universal gravitation, the force of mutual attraction of two bodies is directly proporitional to the product of their masses and inversely proporitonal to the square of the distance between them:

$$F = f\frac{Mm}{r^2},$$

where M is the mass of the first body; m is the mass of the second body; r is the distance between them; f is a factor of proportionality, called the gravitational constant; in the C.G.S. sytem, $f = 6.67 \cdot 10^{-8}\,\mathrm{cm}^3/\mathrm{g} \cdot \mathrm{sec}^2$.

The law of universal gravitation made it possible to predict with unprecedented accuracy even the smallest peculiarities in the motion, not only of celestial bodies of the solar system, but also of twin and multiple stars. Only one case of undoubted discrepancy remained for some time a riddle, for the observations of the annual motion of the perihelion of Mercury yielded a value approximately 0.4" larger than the theoretical. However, even this descrepancy was later explained.

We now know the laws of mutual attraction of celestial bodies, but the nature of the gravitational forces remains to this day largely unexplained. Newton made the assumption that gravitation is transmitted mechanically from one body to another by means of a special medium, "ether", which fills the whole space between bodies and penetrates into all bodies; but this assumption and the theory later proposed on mechanical transmission of the action of some bodies on others could not satisfactorily explain all the features of gravitation.

One fact in particular remained unexplained: that gravitation appears instantaneously; as soon as the bodies change position, the forces acting on the bodies vary immediately.

If gravitation is transmitted mechanically by a medium, then, naturally, the assumption of a finite velocity of propagation of gravitation arises. However, in the observed motions of celestial bodies no deviations from the calculated position due to a finite velocity of propagation of gravitation has been detected.

Laplace attempted to calculate the miminum propagation velocity of gravitation, when the corrections for the finite velocity are so small that they cannot be observed in the motion of celestial bodies. It turned out that in this case the propagation velocity of gravitation should exceed the velocity of light by at least a factor of 1 million. Thus, mechanical theories cannot explain the nature of this rotation.

The General Theory of Relativity developed by the outstanding scientist Albert Einstein brought about profound changes in the concept of space and time, of the motion of material bodies, and of gravitation. This was the next step in the knowledge of the laws and nature of gravitation.

In classical mechanics, the basic laws of which were formulated by Galileo and Newton, there is no mutual relation between space, time, and the physical bodies in the space. Any process taking place in space is measured by time, which flows rigorously, uniformly, and independently. The properties of space remain unchanged and independent of the presence of physical bodies in it. In this space, the Euclidian geometry is true: the shortest distance between two points is a straight line; light rays propagate along a straight line; the geometrical dimensions of bodies and their form are unchanged. The mass of bodies is also constant.

The Theory of Relativity shows that these concepts only approximately reflect reality and are sufficiently accurate only if the velocity of the bodies is low compared with the velocity of light. In reality, a profound mutual relation exists in nature between space, time, and matter.

The path of a body moving by inertia in space is curved, as is the propagation of light rays. The curvature of the path is determined by the attraction of celestial bodies. Thus, the geometry of space is determined by the positions of the physical bodies in it.

Space is homogeneous only if material bodies are absent from it. The real nonuniformity of space, its "curvature," is perceived as gravitation. According to the Theory of Relativity, gravitation is simply a manifestation of the space-time character of the world.

The laws of gravitation following from the General Theory of Relativity reduce to Newton's law as the ratio of the body's velocity to the velocity of light decreases.

The assumptions of the General Theory of Relativity have been brilliantly confirmed. Curvature of light rays when passing near material bodies has been observed. Exact measurements of the position of stars near the edge of the solar disc, made during solar eclipses, show that the stars are not found at their usual places, but are displaced towards the sun by approximately 2". The new relativistic theory of gravitation shows that the perihelion of the planetary orbits, in addition to the displacements caused by Newtonian attraction, should be displaced in each revolution of the planet by a fraction of a revolution equal to $3v^2/c$, where v is the velocity of the planet, and c is the velocity of light.

For Venus, the Earth, and Mars, due to their relatively low orbital velocities, these annual perihelion displacements are very small (for Venus 0.086', for the Earth, 0.039", for Mars 0.014"), and have therefore not been detected by observations. An accurate determination of the perihelion of these planets is difficult because of the small eccentricity of their orbits. For Mercury, the annual displacement of the perihelion is 0.43", i.e., the magnitude obtained from observations, but not accounted for by Newton's law.

It is clear that even the General Theory of Relativity, like any theory explaining certain natural phenomena, cannot account for and describe all of their diversity and infinite interconnections. The law of universal gravitation, formulated by Newton, is the first step in the study of the interaction of bodies. Einstein's theory is the next step in knowledge of the truth.

§2. Unperturbed Motion of Spaceships

The motion of spaceships, as well as of other celestial bodies, is due to the mutual attraction of bodies, expressed by the law of universal gravitation, as was mentioned previously.

If a body m has a unit mass, then a point body with a mass M acts on it with a force

$$V = \frac{\mu}{r},$$

where $\mu = fM$. The force V is called the Newtonian potential of the body M.

As seen from the formula, the larger the mass of the body M the larger (for a given value of r) the Newtonian potential. Thus, the Newtonian potential of the sun is larger than that of the earth, and the potential of the earth is larger than that of the moon.

The Newtonian potential of a celestial body at some point of space is equal to the work which must be done in order to overcome the force of attraction in moving a body of unit mass from the given point to infinity.

The determination of the potential of real celestial bodies having a complicated form and distribution of masses is difficult. Only those celestial bodies which have a spherical form and a uniform density distribution at all points possess the same potential as do point bodies with equal masses. Hence it may be concluded that a spherical celestial body with a uniform density distribution attracts a material point in space as though the entire mass of the body was concentrated in its center. This assumption is used when determining the laws of unperturbed motion of spaceships.

Thus, we will assume that a real celestial body (the sun, the earth, the moon, and so on) has mass M, which is concentrated in its center, and that the mass of the spaceship moving in space is equal to m. The forces of mutual gravitation impart to the ship an acceleration j, with respect to the celestial body whose components along the axes of a rectangular coordinate system with origin at the center of the celestial body, and stationary with respect to the stars, will be (Figure 25):

$$j_x = \frac{d^2x}{dt^2} = -\mu\frac{x}{r^3};$$
$$j_y = \frac{d^2y}{dt^2} = -\mu\frac{y}{r^3};$$
$$j_z = \frac{d^2z}{dt^2} = -\mu\frac{z}{r^3}.$$

These differential equations describe the unperturbed motion of a spaceship. Their solution, omitted here, leads to the following results:*

$$C_1x + C_2y + C_3z = 0;$$
$$r^2\frac{\Delta\varphi}{\Delta t} = C_3;$$
$$r = \frac{p}{1 + e\cos\vartheta},$$

where C_1, C_2, C_3, p, and e are constants.

* The complete solution of the two-body problem can be found in all texts on celestial mechanics. See, for example, /11/ and /31/.

The first of these equations is that of a plane passing through the origin of the coordinates. Consequently, the orbit of the spaceship in unperturbed motion describes a plane curve, stationary with respect to the stars and lying in a plane which passes through the center of the celestial body.

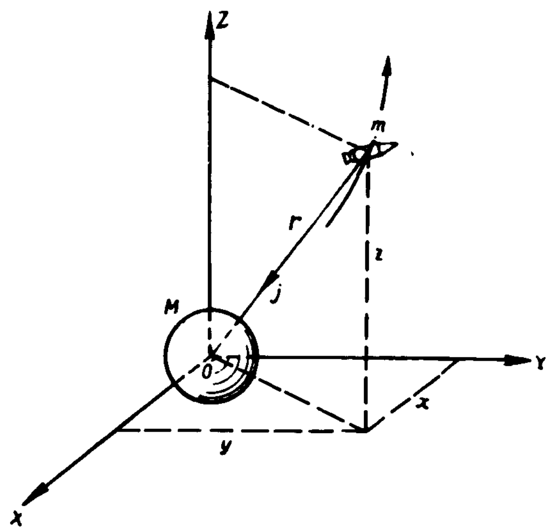

FIGURE 25. Acceleration of a spaceship in the gravitational field of a celestial body with mass M

In the second equation $\Delta\varphi$ is the rotation angle of the radius-vector r during the time interval Δt and therefore the arc of the orbit of a spaceship, corresponding to the time interval Δt, will be $r\Delta\varphi$. The area of an elementary triangle described by the radius-vector during this time interval is $0.5\ r\cdot r\Delta\varphi$. Dividing this area by Δt we obtain the sectorial velocity of the spaceship

$$v_{\text{sect}} = 0.5r^2\frac{\Delta\varphi}{\Delta t}.$$

Comparing the value for the sectorial velocity with the second equation, we note that the left-hand side is twice the sectorial velocity. Hence it follows that the sectorial velocity of a spaceship, or the area swept by its radius-vector, in a unit of time, is constant for all points on its orbit (Kepler's second law).

The third equation is the equation of a second-order curve, or of a conic section in a polar coordinate system with a focus at the center of the celestial body and polar axis directed from the focus to the nearest vertex of the curve. The parameter of this curve is $p = \frac{C_3^2}{\mu}$, the eccentricity $e = \sqrt{1 + \frac{C_4 C_3^2}{\mu^2}}$ and the polar angle $\vartheta = \varphi - \varphi_0$ is the true anomaly of the spaceship ($C_4 = \mu/a$ is an integration constant).

As is known, the ellipse (which in a particular case reduces to a circle), the parabola, and the hyperbola are conic sections. For an ellipse, $e < 1$, for a parabola, $e = 1$, and for a hyperbola, $e > 1$. Thus, a spaceship moves either along an elliptical, parabolic, or hyperbolic orbit (Figure 26). Motion is also possible along a circular orbit, in which case $e = 0$, and along a radial orbit.

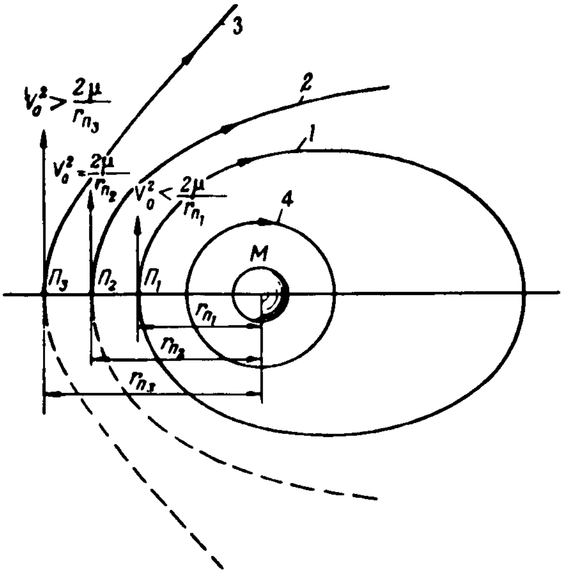

FIGURE 26. Types of orbits of spaceships

1- elliptical; 2- parabolic; 3- hyperbolic; 4- circular; P_1, P_2, P_3, - perigees (perihelions) of the orbits.

The form of the orbit of the spaceship is determined by its velocity V_0 at the nearest vertex (perigee or perihelion). If $V_0^2 < \frac{2\mu}{r_p}$, then the ship moves along an elliptical orbit. For $V_0^2 = \frac{2\mu}{r_p}$ the motion will be along a parabolic orbit, and for $V_0^2 > \frac{2\mu}{r_p}$, along a hyperbolic orbit. Consequently, the form of the orbit of a spaceship is determined by its linear velocity at perigee or perihelion. Conversely, for the spaceship to move along a prescribed orbit, it should have a certain velocity, which depends on the mass of the celestial body, on the distance of the perigee or perihelion, and on the required form of the orbit.

Let us now imagine two spaceships moving along different elliptical orbits around the same celestial body. Denoting their periods of revolution by P_1 and P_2 and the semimajor axes of the orbits respectively by a_1 and a_2, it is possible to obtain the formula

$$\frac{P_1^2}{P_2^2} = \frac{a_1^3}{a_2^3}.$$

This means that for motion along elliptical orbits, the squares of the periods of revolution are proportional to the cubes of the semimajor axes of their orbits (Kepler's third law). It follows from this formula that in the case of motion along an elliptical orbit, the period of revolution of a spaceship is independent of the eccentricity of the orbit, and is determined solely by the magnitude of the semimajor axis. Therefore, spaceships moving along elliptical orbits with different eccentricities, but with semimajor axes of identical length, have equal periods of revolution. Such elliptical orbits with equal semimajor axes and different eccentricities are called equal-energy ellipses.

These are the laws of unperturbed motion of spaceships. We repeat once more that these laws are also correct for natural celestial bodies such as planets and their satellites, comets, meteors, and so on.

§3. Cosmic Velocities

One of the basic motion parameters of a spaceship is its velocity. The magnitude of the velocity not only determines the duration of the "flight" from one point of space to another, which is important in space navigation, but also, as shown above, determines the form of the orbit. We shall therefore consider this problem in more detail.

The solution of the differential equations describing the motion of a spaceship also leads to the following formula, determining its velocity at any point of the orbit:

$$V^2 = \frac{2\mu}{r} + C_4,$$

where $C_4 = \frac{\mu(e^2-1)}{p}$.

From analytic geometry it is known that for an elliptical orbit the square of the eccentricity is

$$e^2 = \frac{a^2 - b^2}{a^2},$$

and the parameter of the orbit is

$$p = \frac{b^2}{a},$$

where a and b are the semimajor and semiminor axes of the orbit, respectively.

Therefore

$$C_4 = -\frac{\mu}{a}.$$

The velocity of the spaceship at a given point of the elliptical orbit is then determined by the formula:

$$V^2 = \mu\left(\frac{2}{r} - \frac{1}{a}\right).$$

This formula can be represented in a different form. Let the radius of the celestial body with respect to which the ship is moving be R, the instantaneous height above its surface be H, and the gravitational acceleration on the surface of the celestial body be g_0.

In this case

$$r=R+H.$$

Let us now imagine that some body with mass m_1 lies on the surface of the celestial body. Its weight will be m_1g_0, but on the basis of the law of universal gravitation

$$m_1g_0=f\frac{Mm_1}{R^2},$$

from which

$$fM=\mu=R^2g_0.$$

Now the formula for the velocity of the spaceship at any point of its elliptical orbit will be

$$V^2=R^2g_0\left(\frac{2}{R+H}-\frac{1}{a}\right).$$

If we consider the motion of a satellite of the earth, then in this formula R is the radius of the earth, g_0 is the gravitational acceleration at the surface of the earth, and H is the altitude of the ship above the surface of the earth at the given moment. If we consider the motion of a spaceship relative to the sun, then in the formula, R, g_0, and H are respectively the radius of the sun, the gravitational acceleration on its surface, and the ship's altitude above the surface of the sun.

It follows from the last formula that the velocity of a spaceship along an elliptical orbit does not remain constant. In the above equation the second term in the brackets, for a given elliptical orbit, as well as the product R^2g_0 are constant, and therefore the maximum velocity corresponds to the smallest value of H, i. e., the perigee (perihelion), and the minimum velocity, to the largest value of H the apogee (aphelion).

As an example, the formula was used to calculate the velocity at two points, perigee (H_p = 226 km) and apogee (H_a = 1881 km), of the third Soviet artificial earth satellite, launched into orbit on 15 May, 1958. The velocity at perigee was found to be about 8.19 km/sec and at apogee, 6.55 km/sec.

A circular orbit is a particular case of the elliptical orbit, and therefore the above formula is also correct for a circular orbit. But when moving along a circular orbit, the semimajor axis $a=R+H$. Substituting this value of a in the previous formula and setting $V=V_{c_1}$ we obtain

$$V_{c_1}=\sqrt{\frac{R^2g_0}{R+H}}.$$

For a given circular orbit the altitude of a spaceship remains constant, and therefore its velocity at all points of a circular orbit is constant.

The velocity determined by the last formula is called circular, and in the case of the earth is the first cosmic velocity.* As can be seen from the formula, the magnitude of the circular velocity depends not only on certain physical characteristics of the celestial body (R, g_0) but also on the height of the ship above the surface. The larger the R and g_0 of the celestial body, the higher the circular velocity, the greater the altitude of the

* This formula may also be easily obtained by using the fact that in the case of motion along a circular orbit the force of inertia is equal to the force of attraction of the spaceship by the celestial body.

spaceship, the lower the velocity which it needs in order to move along a circular orbit (Figure 27).

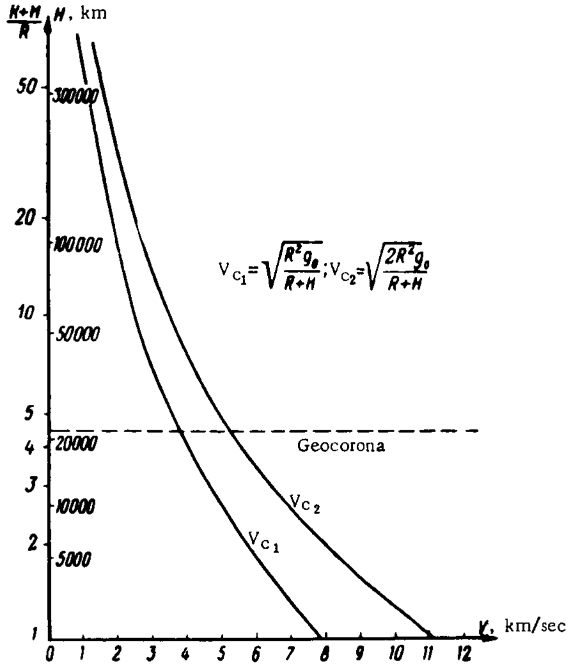

FIGURE 27. Dependence of the first and second cosmic velocities on the altitude of the spaceship

The velocity of a spaceship for $H = 0$ is sometimes called zero circular velocity. Zero circular velocities for various celestial bodies were given in the first chapter.

At a distance of 384,400 km from the earth, which is equal to the mean distance between the moon and the earth, the first cosmic velocity is equal to 1.02 km/sec. This is the mean velocity of the moon along its orbit. For a satellite of the sun the circular velocity at the distance of the earth's orbit from the sun is equal to the mean velocity of the earth around the sun, 29.8 km/sec.

Thus, for a spaceship to move along a circular orbit at a given altitude, it is necessary to give it circular velocity at this altitude. This is not the only condition. The second is that the instantaneous velocity vector be perpendicular to the radius-vector of the ship. Nonfulfillment of one of these conditions results in elliptical motion. It is interesting to note in this connection that if several bodies are launched with circular velocity from one point of space in different directions, after some time all these bodies return simultaneously to the starting point.* This is due to the fact that they will move along elliptical equal-energy orbits with one common point, for which, as was indicated above, the periods of revolution are equal.

* With the exception, of course, of those which in their motion meet the surface of the celestial body.

The circular velocity is not, as is sometimes thought, the minimum velocity for the existence of a satellite of a celestial body. Suppose that the satellite receives circular velocity at point A (Figure 28). It will then move along the circular orbit *1*. However, it can also move along the elliptical orbit *2*, where the velocity required at point A is evidently lower. Finally, motion along orbit *3* is also possible in principle. In this case, an even lower velocity is required at point A. Let us confirm this by calculation.

Let us assume that the point A is situated 500 km above the surface of the earth. The first cosmic velocity at this height is

$$V_1=\sqrt{\frac{R^2g_0}{R+H}}=\sqrt{\frac{6371^2\cdot 981\cdot 10^{-5}}{6371+500}}=7.62\ \text{km/sec}.$$

This velocity results in motion along the circular orbit *1*. If the height of the point P_2 is equal, for example, to 150 km, and of the point P_3 is equal to 0 km, then the lengths of the semimajor axes for the second and third orbits are:

$$a_2=\frac{2\cdot 6371+500+150}{2}=6696\ \text{km};$$

$$a_3=\frac{2\cdot 6371+500+0}{2}=6621\ \text{km}.$$

Then the velocities required at the point A in order to move along these orbits will be:

$$V_2=\sqrt{R^2g_0\left(\frac{2}{R+H}-\frac{1}{a_2}\right)}=7.51\ \text{km/sec};$$

$$V_3=\sqrt{R^2g_0\left(\frac{2}{R+H}-\frac{1}{a_3}\right)}=7.47\ \text{km/sec}.$$

For motion along a parabolic orbit $e=1$ and therefore $C_4=0$. Taking $V=V_{c_2}$, we obtain

$$V_{c_2}^2=\frac{2\mu}{r},$$

or, analogously to the previous formulas,

$$V_{c_2}=\sqrt{\frac{2R^2g_0}{R+H}}.$$

The velocity determined from this formula for the earth is called the second cosmic velocity. For all celestial bodies this velocity is also called parabolic, and sometimes escape velocity* or break-off velocity, meaning "escape" and "break-off" from the action of the gravitational field of the celestial body. In fact, a parabola is a curve whose branches go to infinity, and therefore a spaceship having at a given point in space the velocity V_{c_2} goes from the celestial body to infinity, i.e., completely overcomes gravitation. As can be seen from the formula, the velocity of the spaceship at infinity ($H=\infty$) will in this case be equal to zero.

The first chapter gives the values of the escape velocity for the celestial bodies of the solar system. Figure 27 shows the dependence of the second cosmic velocity on the height above the earth's surface.

* The formula for the escape velocity can also be derived on the basis of the Law of Conservation of Energy. The kinetic energy of the spaceship should be equal to the work expended to move it in the gravitational field of the celestial body from the given level H to infinity.

Thus, velocities lower than the escape velocity correspond to elliptical orbits, and the escape velocity, to parabolic orbits. It is natural now to draw the following conclusion: the closer the velocity of a vehicle to the escape velocity, the more elongated the elliptic orbit will be, and the further from the celestial body it will move.

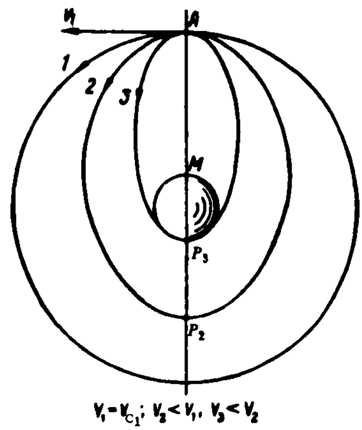

FIGURE 28. Minimum velocities of a satellite ship for flight along 1) circular and 2) and 3) elliptical orbits

Sub-parabolic velocities are characteristic of satellites of celestial bodies, and parabolic and super-parabolic velocities correspond to flights beyond the gravitational boundaries of a given celestial body, for example, interplanetary flights from the earth.

Comparing the formulas for the circular velocity and for the escape velocity, one observes the simple relation:

$$V_{c_2}=V_{c_1}\sqrt{2}\approx 1.41V_{c_1}$$

As was already indicated, velocities higher than the escape velocity are called hyperbolic. A spaceship with a hyperbolic velocity goes to infinity like the case of escape velocity, but the curvature of the orbit is smaller, the ship departs more rapidly from the celestial body, and the velocity at infinity is higher than zero.

For a hyperbola

$$e^2=\frac{a^2+b^2}{a^2};$$

$$p=\frac{b^2}{a};$$

$$C_4=\frac{\mu}{a}$$

The formula for the velocity in this case takes the form:

$$V^2=\mu\left(\frac{2}{r}+\frac{1}{a}\right)$$

or

$$V^2=R^2g_0\left(\frac{2}{R+H}+\frac{1}{a}\right).$$

This formula determines the velocity of the spaceship at any point of a hyperbolic orbit. As can be seen from the formula, at infinity $(H = \infty)$

$$V^2 = \frac{R^2 g_0}{a}.$$

Let us determine the velocity which, at the mean distance of the earth from the sun, ensures the escape of a spaceship from the gravitational field of the sun. It is clear that its magnitude can be calculated by the formula for the escape velocity, if we put R = the radius of the sun, g_0 = the gravitational acceleration at the surface of the sun, and $R + H$ = the mean distance between earth and sun.

This velocity can also be obtained in another way. The circular velocity of a satellite of the sun at the distance of the earth's orbit is equal to the mean velocity V_e of the earth along its orbit around the sun. Knowing the relation between the circular velocity and the escape velocity, we obtain

$$V = V_e \sqrt{2} = 29.8 \cdot 1.41 = 42.1 \text{ km/sec}.$$

Thus, the heliocentric velocity, the escape velocity of a spaceship with respect to the sun is 42.1 km/sec. Determination of the escape velocity of the ship with respect to the earth, the geocentric velocity, is an important problem since the spaceship must take off from the earth.

Calculations show that the minimum velocity with respect to the earth of a spaceship to start from the surface of the earth for flights beyond the limits of the solar system should be 16.7 km/sec. This constant velocity is called the third cosmic velocity. We will discuss this and show its physical meaning in a later section. Now we only note that this velocity is not high, if we compare it with the second cosmic velocity (11.2 km/sec at the surface of the earth). This indicates that currently existing rockets will in principle be able to fly beyond the limits of the solar system.

Of course, spacecraft can also move with respect to celestial bodies with higher velocities — reaching nearly the velocity of light.*

In concluding this section we consider an important parameter of motion, the revolution period P of a satellite around a celestial body.

Consider a satellite moving around a celestial body of radius R along a circular orbit at a height H. Obviously,

$$P = \frac{2\pi (R+H)}{V_{c_1}} = \frac{2\pi (R+H)}{\sqrt{\frac{R^2 g_0}{R+H}}} = \frac{2\pi}{R} \sqrt{\frac{(R+H)^3}{g_0}}.$$

But for a satellite moving along a circular orbit, $R + H = a$, and therefore,

$$P = \frac{2\pi}{R} \sqrt{\frac{a^3}{g_0}},$$

where a is the semimajor axis of the orbit.

It was shown above that the period of revolution is independent of the eccentricity of the orbit. Ships moving along orbits with equal semimajor axes have equal periods of revolution. Consequently, this formula is also true for ships moving along elliptical orbits, if the semimajor axis of the

* Measurements of the velocity of light made in 1951 resulted in the value 299,792 km/sec, with a probable accuracy of 1 to 2 km/sec.

elliptical orbit is calculated by the formula

$$a=R+\frac{H_p+H_a}{2},$$

where H_a and H_p are respectively the heights of the furthest (apogee, aphelion) and nearest (perigee, perihelion) points of the orbit. The revolution period also depends on such physical characteristics of the celestial body as its radius and the gravitational acceleration at the surface (Tables 5 and 6).

TABLE 5. Revolution periods of a satellite ship orbiting the earth

Height of circular orbit, km	Semimajor axis, km	Period of revolution	Height of circular orbit, km	Semimajor axis, km	Period of revolution
0	6370	$1^{hr}24.4^{min}$	5000	11,370	$3^{hr}22^{min}$
100	6470	1 27	10,000	16,370	5 49
200	6570	1 29	20,000	26,370	11 30
300	6670	1 31	30,000	36,370	18 37
500	6870	1 35	35,870	42,240	24 00
1000	7370	1 46	40,000	46,370	27 44
2000	8370	2 08	50,000	56,370	37 11
3000	9370	2 31	100,000	106,370	96 24
				384,400*	27.32 days

* The mean distance from the earth to the moon.

TABLE 6. Revolution periods of satellite ships orbiting the planets and the moon

Planet	Revolution period, hours		
	H = 0 km	H = 1000 km	H = 5000 km
Mercury	1.48	2.46	7.70
Venus	1.49	1.88	3.63
Mars	1.61	2.38	6.38
Jupiter	2.86	2.92	3.18
Saturn	3.92	4.01	4.43
Uranus	2.95	3.13	3.85
Neptune	2.63	2.80	3.46
Moon	1.82	3.60	13.90

It follows from Table 5 that the revolution period of a satellite orbiting the earth, in a circular orbit at a height of H = 0, is equal to 84.4 min. As is known, this is also the oscillation period of a Schuler pendulum, a hypothetical pendulum with a plumb-line equal in length to the radius of the earth and with the point of suspension at the surface of the earth. The remarkable property of such a pendulum is its unperturbability. Acceleration applied to such a pendulum does not move it out of the vertical. This property of the Schuler pendulum is widely used in inertial navigation systems of aircraft and ships, which are essentially physical analogues of a hypothetical pendulum with an oscillation period of 84.4 min.

The launching of a satellite which is synchronous with respect to the surface of the earth is also of interest. This satellite should obviously have a revolution period of 24 hours. This, however, is not the only condition. The plane of any satellite's orbit coincides with the center of the earth, and therefore a synchronous satellite is only possible if the inclination of its orbit $i = 0°$. In other words, the satellite should be launched over the equator in the direction of the earth's rotation (from west to east) into a circular orbit 35,870 km above the surface of the earth with an orbital velocity of approximately 3.07 km/sec (see Table 5). Such a satellite will "hang" above the same point of the equator. Some scientists foresee a wide use of such satellites in world television systems, navigation, and communication systems.

§4. Perturbations of the Orbits of Spaceships

All the foregoing considerations were devoted to unperturbed, or Keplerian motion of spaceships, whose laws are determined, as was shown above, by the solution of the two-body problem. However, Keplerian motion is only the first approximation to the true motion of spacecraft.

The true motion of spaceships takes place along orbits of a more complicated form. The main reasons for this in the case of low-orbit artificial earth satellites is the influence of the atmosphere and the nonuniform nature of gravity due to the flattening of the earth at the poles and the nonuniform distribution of its mass.

The influence of the gravitational fields of the sun, moon, and planets on the motion of low-orbit satellites is actually slight but in interplanetary flight they become important, and at certain distances overcome the effect of the earth's gravitational field.

In general the perturbations are comparatively small, and the difference between the actual and unperturbed orbits is small. It is therefore assumed that at each moment, the motion is along an unperturbed orbit, whose elements continuously vary as a result of the perturbations. Such orbits with time-varying elements are called oscillating orbits.

The orbit perturbations are divided into secular and periodical. The secular perturbations are characterized by a continuous variation of the orbital elements in one direction, and the periodical perturbations are characterized by an oscillation of the orbital elements about some mean position.

Let us consider the perturbations of the orbits of low-orbit satellites in greater detail. Periodical perturbations due to the nonuniform nature of the earth's gravitational field give rise to deviations, of about 10 km, of the true orbits of the satellites from the unperturbed orbits.

The main secular perturbation of the orbits of satellite ships due to the nonuniform nature of gravity, is the precession of the orbit, which is manifested in a slow rotation of the plane of the orbit with respect to the axis of the earth (Figure 29). The inclination of the orbit in this case remains practically unchanged.

The mean velocity of angular precession ω_p of the orbit is given by the formula

$$\omega_p = \frac{2\pi R_e^2}{Pp^2}\left(\alpha - \frac{\omega_E^2 R_e}{2g_0}\right)\cos i,$$

where R_e is the equatorial radius of the earth; g_0 is the gravitational acceleration at the equator; α is the flattening of the earth; ω_E is the velocity of angular rotation of the earth; i is the inclination of the orbit; p is the parameter of the orbit; P is the rotation period of the satellite.

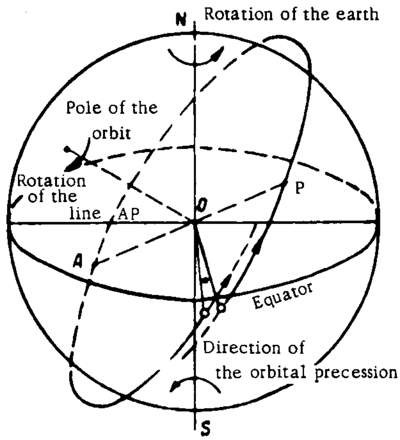

FIGURE 29. Precession of the orbit of an earth satellite due to the nonuniform nature of gravity (orbit inclination 65°)

As seen from the formula, the velocity of angular precession depends on a number of parameters, including the inclination of the orbit (Table 7). For satellites on polar orbits (i = 90°) the angular precession velocity of the orbit is zero.

TABLE 7. Precession velocities and rotation velocity of the line of apsides of the orbits of the first, second, and third Soviet artificial earth satellites at the beginning of their existence

Artificial earth satellite	Inclination of the orbit, degrees	Revolution period, min	Velocity of precession of the orbit, deg/days	Velocity of rotation of the line of apsides of the orbit, day
First	65.129	96.17	3.157	0.432
Second	65.310	103.75	2.663	0.407
Third	65.188	105.95	2.528	0.326

Thus, for the first Soviet artificial earth satellites the precession of the orbit per revolution was approximately 15'.

The nonuniform nature of the earth's gravitational field is the reason for another secular perturbation of the orbits of satellites. This is manifested in the rotation of the line of apsides in the plane of the orbit, as a result of which the angular distance between the perigee and the ascending node continuously decreases. The latitude of the geographical location of perigee and of apogee accordingly varies. As can be seen from Table 7, this perturbation for the first Soviet artificial satellites was considerably smaller than the precession of the orbit.

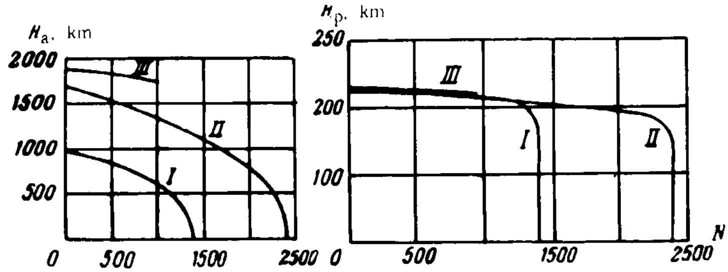

FIGURE 30. Dependence of the height of the apogee H_a and the perigee H_p of the first (I), second (II), and third (III) Soviet artificial satellites on the number of orbits around the earth

Thus, the nonuniform nature of gravity changes the orientation of the ship's orbit in space but does not affect the form of the orbit. Another source of secular perturbations, atmospheric drag, has very little effect on the orientation of the orbit, but considerably modifies its form.

The atmospheric drag reduces the energy of the satellite, resulting in a decrease of its flight altitude (Figure 30). Analyzing the observed data of the first artificial earth satellites, it can be seen that until penetration into the dense layers of the atmosphere ($H \approx 200$ km), the decrease in the height of the apogee is faster than the decrease in the height of the perigee. Consequently, due to the atmospheric drag, the elliptical orbit of a satellite approaches a circular form.

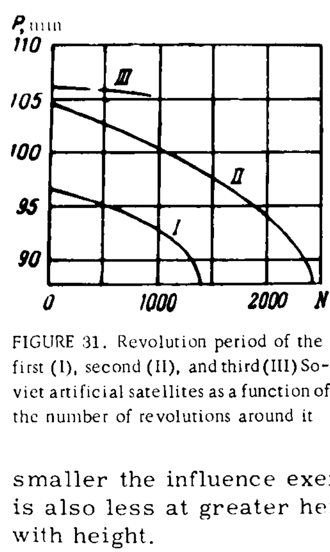

FIGURE 31. Revolution period of the first (I), second (II), and third (III) Soviet artificial satellites as a function of the number of revolutions around it

Due to the lowering of the flight altitude, the revolution period of the satellite decreases (Figure 31). The rate of deceleration of a satellite depends on its cross-sectional load, the ratio of the weight of the satellite to the area of its (middle) cross section, and on a coefficient determined by the geometrical form of the satellite and its orientation in space. The higher the cross-sectional load, the smaller the influence exerted by the atmosphere on the satellite. The drag is also less at greater heights due to the decrease in atmospheric density with height.

Atmospheric drag also reduces the altitude of a satellite in a circular orbit (Table 8).

Above an altitude of 140 to 160 km, the satellite makes altogether only 1 to 2 revolutions.

The duration of satellite's motion from the moment of being put into orbit to complete braking in the dense layers of the atmosphere is called the satellite's lifetime. This depends not only on the cross-sectional load, but to a considerable extent also on its flight altitude. With increasing altitude the lifetime increases. Thus, for circular orbits, an increase in altitude from 250 km to 300 km increases the lifetime by a factor of 5; from 300 to 400 km, by a factor of 8; and from 400 to 500 km, by a factor of approximately 6.3 (Table 9).

TABLE 8. Decrease in the height of a circular orbit of an earth satellite in one revolution (average data for one of the laws of variation of the atmospheric density) (Cross-sectional load about 20 kg/m²)

Orbit height, km	480	400	320	240	190	160
Decrease in the height per one revolution, km	0.016	0.032	0.145	1.05	7.6	53.1

TABLE 9. Lifetimes of an earth satellite (weight 100 kg, diameter 1 m, cross-sectional load about 128 kg/m²)

a) circular orbit

Orbit height, km	200	250	300	350	400	500
Lifetime, days	0.4	4	20	65	160	1010

b) elliptical orbit (lifetime in days)

Height of perigee, km	Height of apogee, km				
	500	700	1000	1300	1600
200	9	18	37	58	82
230	25	52	102	165	237
260	53	116	238	370	535
300	114	260	545	890	1280
400	410	1120	2630	4450	6600

The lifetime of satellites is directly proportional to the cross-sectional load and inversely proportional to the drag coefficient. Thus, for a satellite with a diameter of 2 m, weighing 1000 kg (cross-sectional load of about 319 kg/m²), the lifetime will be longer than those given in Table 9 by a factor of approximately 2.5 /2/.

As was already said, the gravitational fields of the sun, the moon, and the planets exert a small influence on the motion of low-orbit satellites. Only at altitudes measured in tens and hundreds of thousands of kilometers do the effects of other celestial bodies, and primarily the sun and the moon, become noticeable in practice.

The perturbing actions of the sun and the moon on a satellite lead first of all to a variation in the height of the perigee. Depending on the position of the orbit with respect to the sun, the height of the perigee may increase or decrease. The variation in one revolution depends on the height of the apogee; the higher the apogee, the more the height of the perigee varies. Decrease in the height of the perigee can considerably shorten the lifetime of a satellite in comparison with the nominal values, which take into account only atmospheric drag.

Thus, for example, the Soviet automatic interplanetary station launched on 4 October, 1959, which photographed the far side of the moon, passed near the earth at a height of somewhat over 40,000 km on its first revolution. The apogee was approximately 480,000 km. At such great heights, atmospheric drag is practically nil and it could therefore be expected that the station would remain in orbit for a very long time. However, the perturbation of the sun caused such a sharp decrease in the perigee, that the lifetime of the station was only about half a year.

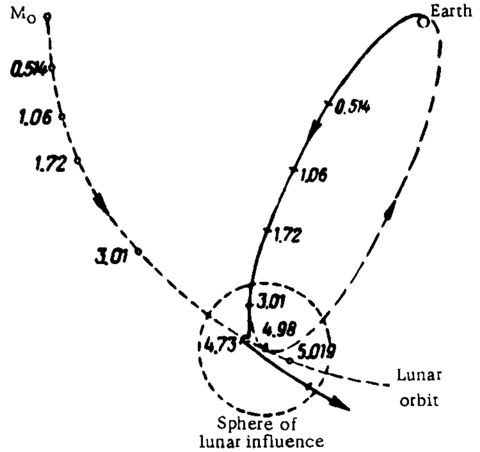

FIGURE 32. Perturbation of the orbit of the spaceship launched to the moon, calculated by V.A. Egorov. The dashed line shows the orbit without allowance for the perturbation of the moon; the figures on the orbits of the ship and of the moon give the duration of the motion in days from the moment of launching

In order to prolong the lifetime of such automatic stations and spaceships three conditions must be satisfied. The apogee should be as small as possible. For example, for a flight to the moon, the apogee should not greatly exceed the distance from earth to moon. Furthermore, the perigee on the first revolution should be sufficiently large. Finally, the moment of launching should be chosen so that the perturbation of the sun does not lead to a decrease in perigee.

The perturbations of the orbit of a spaceship are particularly large if it enters the gravitational field of another celestial body (Figures 32 and 33).

Perturbations also arise from the pressure of solar rays on the spaceship. However, such perturbations are observable only in the case of long flights relatively near to the sun.*

Thus, flights to the moon and to other planets require calculation of the orbits which take into account the perturbations in the gravitational fields of these celestial bodies.

* [The effects of solar pressure are inversely proportional to the cross-sectional load. Thus, the American "Echo" satellite, a hollow sphere of extremely low cross-sectional load, was seriously affected by solar pressure.]

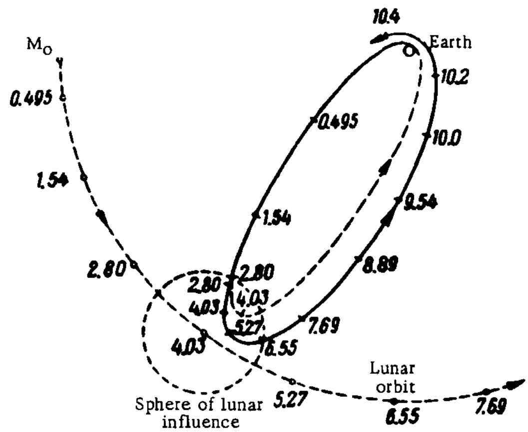

FIGURE 33. Perturbation of the orbit of a spaceship with a calculated apogee at the point where the gravitational attractions of earth and moon are equal, calculated by V.A.Egorov. The dashed line shows the orbit without allowance for lunar perturbations; the figures on the orbits of the ship and moon give the duration of the motion in days from the moment of launching

§5. Zone of Predominance of the Attraction and the Sphere of Action of the Earth and of Other Celestial Bodies

According to the law of universal gravitation, the force of attraction is inversely proportional to the square of the distance. Consequently, as one gets further from a celestial body the force of attraction acting on the ship rapidly decreases. But the force of attraction is proportional to the product of the mass of the celestial body and the mass of the spaceship. Therefore, in general, two celestial bodies, equidistant from a spaceship, act unequally on it. Obviously, there will be a series of points in space at which the attraction of the ship by both celestial bodies is equal. These points form some surface in space, which is the boundary of the region of predominance of the attraction of one celestial body over the other (Figure 34).

We will now give the solution of a particular problem — to find the point on the straight line connecting the centers of two celestial bodies, at which their attraction is equal.

The attraction of the ship by the two bodies is:

$$F_1 = f\frac{M_1 m}{r_1^2}; \qquad F_2 = f\frac{M_2 m}{r_2^2},$$

where M_1 and M_2 are the masses of the celestial bodies, and r_1 and r_2 are the distances from their centers to the spaceship.

From the conditions of the problem $F_1 = F_2$, and consequently:

$$\frac{r_1^2}{r_2^2} = \frac{M_1}{M_2} \quad \text{or} \quad \frac{r_1}{r_2} = \sqrt{\frac{M_1}{M_2}}.$$

If the first body is the sun, and the second body is the earth, then $M_{\odot}/M_{\oplus} = 333{,}434$ and $r_{\odot}/r_{\oplus} = 577.432$. The point of equal attractions is situated at a distance of 149,198,620 km from the sun and 258,380 km from the earth.

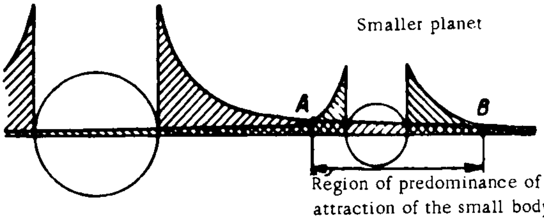

FIGURE 34.* Region of predominance of the attraction of celestial bodies (at the points A and B the attraction of the spaceship by the two bodies is equal)

For the earth-moon system, the point of equal attractions is 38,321 km from the moon and 346,079 km from the earth.

However, knowledge of the region of attraction predominance is not as important in space flights as the knowledge of the parameters of the sphere of action of the celestial body.

Consider two celestial bodies, for example the sun and the earth. The spaceship starts from the earth, flying to other celestial bodies. Since it must have a velocity not less than the second cosmic velocity, it will initially move along a parabolic or hyperbolic orbit with respect to the earth, influenced by the earth's gravitational field. Afterwards, it will follow some orbit under the action of the gravitational field of the sun.

That region of space, within which the orbit of a spaceship is affected by the gravitational field of a given celestial body, is called the sphere of action of the celestial body. It is possible to give a more rigorous definition of this sphere. For example, for the sun-earth system, the sphere of action of the earth is that region of space, within which the ratio of the perturbing force of the sun to the force of attraction exerted by the earth on the spaceship is smaller than the ratio of the perturbing force of the earth to the force of attraction of the spaceship by the sun:

$$\frac{F_{\text{pert.S}}}{F_{E\to s}} < \frac{F_{\text{pert. E}}}{F_{S\to s}},$$

or

$$\frac{F_{S\to s} - F_{S\to E}}{F_{E\to s}} < \frac{F_{E\to S} - F_{E\to s}}{F_{S\to s}},$$

where $F_{S\to s}$ is the force of attraction of the spaceship by the sun; $F_{E\to s}$ is the

* [We may consider the case of a line passing through both celestial bodies. The magnitude of the gravitational attraction of each is shown as the height of the shaded area in Figure 34. The region of predominance of the attractive force of a heavenly body is where its shaded area is higher than the other. Points A and B denote the places where the heights, and therefore the gravitational forces, are equal. The region of predominance of the smaller planet is between points A and B; on either side of it the gravitational attraction of the larger planet is greater.]

force of attraction of the spaceship by the earth; $F_{S \to E}$ is the force of attraction of the earth by the sun; $F_{E \to S}$ is the force of attraction of the sun by the earth.

According to the law of universal gravitation, the last two forces are equal.

The radius of the sphere of action of one celestial body to another of larger mass is given by the formula

$$r_o = r\left(\frac{m}{M}\right)^{\frac{2}{5}},$$

where m and M are the respective masses of the two bodies and r is the distance between them.

The radii of the spheres of action of the planets with respect to the sun have been calculated by this formula (Table 10).

TABLE 10. Radii of the spheres of action for the sun-planet systems

Planet	Radius of the sphere of action, thousands of km	Planet	Radius of the sphere of action, thousands of km
Mercury	110	Jupiter	48,500
Venus	620	Saturn	54,500
Earth	930	Uranus	52,000
Mars	580	Neptune	87,500

The greater the ratio m / M and the distance r, the greater, as can be seen from the formula, is the radius of the sphere of the action of the first body. This explains the large radii of the spheres of action of the remote giant planets.

Within the sphere of action of a planet, the motion of a spaceship is determined mainly by its gravitational field. The gravitational fields of other celestial bodies, including the sun, cause relatively small perturbations of this motion. Outside the sphere of action of the planet, the motion is determined by the gravitational field of the sun; the gravitational fields of the planets cause only small perturbations. When a spaceship flies from the earth to the moon and back (the radius of the sphere of action of the moon with respect to the earth is approximately 66,000 km), the motion within the moon's sphere of action is governed by the moon's gravitational field, and outside of it by that of the earth (Figure 35). The sphere of action of the earth is considerably larger than the zone of predominance of the earth's attraction, as seen from the figure. An interplanetary spaceship will be "tied to the earth" through its gravitational field up to 930,000 km.

Motion within the sphere of action of the earth, considered with respect to the earth, we will call geocentric; motion in the sphere of action of the moon, considered with respect to the moon, we will call selenocentric; motion outside the spheres of action of the earth, of other planets, and of the moon we will call heliocentric.

Calculation of the trajectory of a spaceship in the general case is related to the solution of a problem for several bodies, for example; earth - ship - moon, or earth - ship - sun - Venus. This problem can be approximately reduced to separate solution of the two-body problems; earth - ship and moon - ship or earth - ship, sun - ship, and Venus - ship.

We return now to the problem of the third cosmic velocity and attempt to determine its magnitude. We recall that the third cosmic velocity is the minimum take-off velocity from the surface of the earth which ensures escape from the solar system. As was indicated previously, the heliocentric escape velocity from solar attraction is 42.1 km/sec, and the orbital velocity of the earth with respect to the sun is 29.8 km/sec. We will assume that the ship is launched from the earth in such a way that, on the boundary of the earth's sphere of action, the vector of its geocentric velocity coincides with the vector of the orbital velocity of the earth. In this case the required geocentric velocity on the boundary of the sphere of action of the earth is

$$V_1 = 42.1 - 29.8 = 12.3 \text{ km/sec}$$

Taking into account the necessity to overcome earth's gravitational field, we obtain on the basis of the law of conservation of energy

$$\frac{mV_{c_3}^2}{2} = \Pi_0 + \frac{mV_1^2}{2}.$$

The kinetic energy spent on accelerating a spaceship, starting from the surface of the earth, should be equal to the sum of the potential energy at the surface of the earth and the kinetic energy of a ship having velocity V_1 at the boundary of the earth's sphere of action.

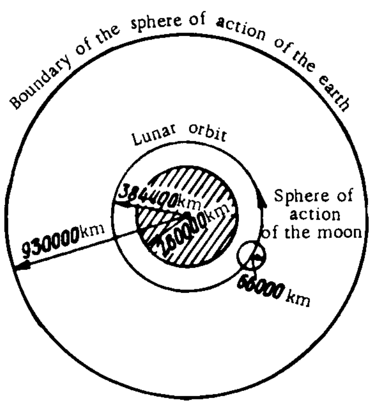

FIGURE 35. Sphere of action of the earth, zone of predominance of the earth's attraction over the sun's attraction (shaded), and the sphere of action of the moon with respect to the earth

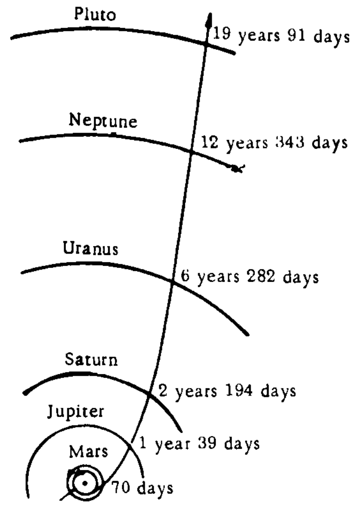

FIGURE 36. Orbit of a ship starting from the earth with the third cosmic velocity

We have already noted that to overcome the gravitational field of the earth, a spaceship starting from the earth should have the second cosmic velocity V_{c_2} = 11.2 km/sec.* Consequently, $\Pi_0 = \frac{mV_{c_2}^2}{2}$ and then $\frac{mV_{c_3}^2}{2} = \frac{mV_{c_2}^2}{2} + \frac{mV_1^2}{2}$. Assuming the mass of the spaceship in flight is constant, we obtain

$$V_{c_3} = \sqrt{V_{c_2}^2 + V_1^2};$$

$$V_{c_3} = \sqrt{11.2^2 + 12.3^2} = 16.7 \text{ km/sec}$$

This is the so-called third cosmic velocity. A star-ship launched with this velocity will first move along a hyperbolic orbit with respect to the earth. Then, after leaving the sphere of action of the earth, it will move along a parabolic orbit with respect to the sun (Figure 36).

Thus, the third cosmic velocity is equal to 16.7 km/sec, but this velocity was obtained without considering the diurnal rotation of the earth. As a result of this rotation, as was shown in the first chapter, all points on the terrestrial surface have a linear velocity $V_r = 0.465 \cos\varphi$ where φ is the latitude of the starting point. Consequently if it is assumed that the vector of this velocity and the vector of the initial velocity of the star-ship coincide, to fly beyond the solar system the ship should have a velocity $V = 16.7 - 0.465 \cos\varphi$. The velocity of the diurnal rotation of the earth should be taken into account in the launching of spaceships of any type.

§6. Trajectories of Interplanetary Ships

Let us consider now the trajectories of interplanetary ships. Assume first that the earth and other planets move along circular orbits whose radii are equal to the mean radii of their true orbits, and that the orbits of the planets and of the spaceships lie in one plane.

In general, the trajectory of an interplanetary ship can be divided into the following phases (Figure 37): launch and vertical flight to satellite orbit height (AB), flight in a circular or elliptical orbit around the earth (BC), flight within the limits of the sphere of action of the earth (CD), flight along an elliptical orbit with respect to the sun (DE), flight within the sphere of action of the destination planet (EF), orbital flight around the destination planet (FG) and, finally, the descent to the surface of the destination planet (GH).

The trajectory points at which the spaceship transfers from one orbit to another are called transition points or orbital transfer points. In general all these points, with the exception of points D and E, a certain amount of energy has to be expended for the transfer of the ship from one orbit to another. At points D and E transfer into the next orbit is made without energy expenditures, since the spaceship passes from the sphere of action of one celestial body to the sphere of action of another.

We shall now describe a typical flight of an interplanetary spaceship. It starts vertically in order to escape more rapidly into the rarer layers of the atmosphere. The flight trajectory then curves according to a certain program (Figure 38), precalculated to give the required altitude H_0 and the

* To reach the boundary of the sphere of action of the earth, a somewhat lower velocity, V = 11.15 km/sec, is necessary.

required inclination angle Θ of the velocity vector to the horizon at the end of the initial section.

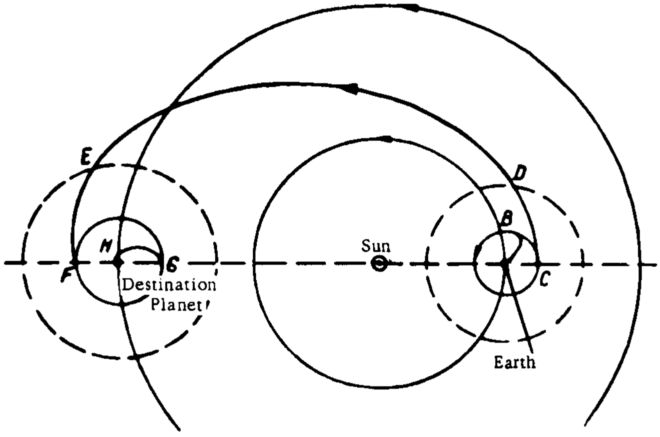

FIGURE 37. Phases of the trajectory of an interplanetary ship

AB - Initial flight; *BC* - circular (elliptical) orbit around the departure planet (earth); *CD* - flight within its sphere of action; *DE* - flight along an elliptical orbit with respect to the sun; *EF* - flight in the sphere of action of the destination planet; *FG* - orbital flight around the destination planet; *GH* - descent to the surface of the destination planet.

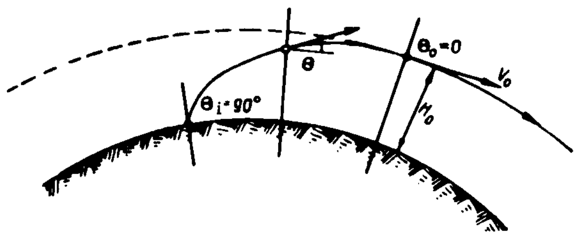

FIGURE 38. Launching a spaceship into its trajectory

Θ - angle of inclination of the velocity vector V to the horizon; H_0 - altitude at the termination of the initial phase.

To obtain a circular orbit, this angle should be zero, and the velocity should be the circular velocity for the given height.

The total velocity which has to be given to the spaceship in order to put it into orbit and overcome the influence of the gravitational field and the atmospheric drag is called the characteristic velocity. The velocity required to compensate for the influence of gravity and the atmospheric drag amounts to about 2 to 3 km/sec. In order for the ship to have a final velocity of 8 km/sec, the ship's propulsion system should possess a characteristic velocity of 10 to 11 km/sec. Similarly, for a final velocity of 11 km/sec, the required characteristic velocity is 13 to 14 km/sec.

Using existing chemical fuels, such characteristic velocities can be attained only by multi-stage rocket systems /2/.

It is possible to calculate the required velocity and angle Θ for an elliptical orbit oriented in any given way. The minimum velocity is required when the end of the intial phase coincides with the apogee. However, to reduce the total energy losses, it is most advantageous to introduce the ship into orbit in the region of its perigee. But since the length of the initial phase is short, this method is inapplicable if the ship has to be introduced into an orbit with a perigee height of several thousand kilometers. In this case the spaceship should first be introduced into a transitional orbit with small perigee (Figure 39). At the apogee it should be given an additional velocity ΔV so that it will move on the preassigned circular orbit around the earth.

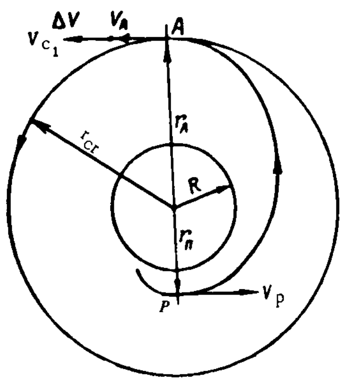

FIGURE 39. Putting a spaceship into a circular orbit from a transitional elliptical orbit

P and A - perigee and apogee of the transitional orbit; $V_{c_1} = V_A + \Delta V$ is the total velocity required for a given orbit.

Calculations have been made for introducing a ship in this way into circular orbits of different heights with the perigee of the transitional orbit kept constant. They show that with increasing height of the preassigned orbit the total velocity required first increases, reaching a maximum value at a circular orbit height of about 100,000 km. It then decreases tending in the limit to the second cosmic velocity (Table 11). Launching of a ship to a greater height requires a lower total velocity, strange as this may seem.

TABLE 11. Velocities at perigee and apogee of the transitional orbit for introducing the ship into a circular orbit of a given height (perigee of the transitional orbit 200 km)

Height of the given orbit, km	Required velocity at perigee of the transitional orbit, km/sec	Additional velocity at apogee of the transitional orbit, km/sec	Total velocity, km/sec
1000	8.009	0.214	8.223
5000	8.769	0.854	9.623
25,000	10.016	1.467	11.483
50,000	10.424	1.444	11.868
75,000	10.595	1.358	11.953
100,000	10.690	1.276	11.966
125,000	10.749	1.204	11.953
150,000	10.790	1.143	11.933
200,000	10.844	1.045	11.889

Calculation of the initial phase should take into account the diurnal rotation of the earth. It is obvious that launching of the spaceship in an easterly direction is more advantageous, since the velocity of the earth's rotation is added to the velocity given the ship by its propulsion system.

The increment in the ship's velocity due to the rotation of the earth largely depends on the inclination of the orbit and the altitude height of the introduction point. It can be approximated by the formula

$$\Delta V = \omega_E (R + H_0) \cos i,$$

where R and ω_E are the radius of the earth and its angular rotational velocity about its axis; H_0 is the height of the introduction point; i is the inclination of the orbit.

The second phase of the trajectory of an interplanetary ship is the flight along a circular or elliptical orbit around the departure planet (Figure 37). This period may be used for various purposes. If the interplanetary ship uses chemical fuel, it may be refueled by "tanker" rockets which join it in orbit. It has even been proposed that large interplanetary rockets be assembled in orbit from subassemblies flown up by "ferry" rockets. This would allow the interplanetary ship to be designed for flight in vacuum only, which would simplify its construction considerably by eliminating parts needed for atmospheric penetration. In any event, the interplanetary ship will wait in orbit for favorable radiation conditions in cosmic space and favorable positions of the departure and destination planets.

From the energy point of view this section is characterized by initial flight, i.e., the interplanetary ship becomes a satellite of the departure planet for a while. At the end of the second phase the ship is given additional velocity for leaving the sphere of action of the departure planet and for further motion along a preassigned elliptical orbit with respect to the sun.

The Soviet space probes launched to Venus on 12 February, 1961 and to Mars on 1 November, 1962 first went into elliptical parking orbits around the earth. The automatic station "Luna IV", which was launched on 2 April, 1963 was also introduced to a trajectory to the moon from an orbit of an artificial satellite of the earth. First the interplanetary stations were introduced into a satellite orbit around the earth, and then into an elliptical orbit with respect to the sun.

The use of a zero-energy elliptical or circular orbit is feasible for all interplanetary ships, but particularly for those with chemical rocket engines. The feature of chemical rockets which makes them suitable is their high thrust. This allows the velocity increment required to transfer from an elliptical orbit around the earth to an orbit with respect to the sun to be generated in a relatively short time.

We may also consider the use of an electron-rocket engine. The best known form of such an engine is the "ion engine." Such an engine has extremely high exhaust nozzle velocities (up to 200 km/sec) and very low thrust. Its acceleration is correspondingly low (approximately 10^{-3}m/sec^2) therefore a spaceship with such an engine cannot transfer from an earth orbit to a solar orbit in a short time. Such a ship would have a long period of acceleration after achieving circular orbit which would take it into increasingly higher circular orbits before achieving solar orbit (Figure 40).

Another engine with a tremendous exhaust velocity and a low thrust is the "photon" engine. It uses the reactive force of a powerful flux of photons and is capable of accelerating a spaceship to tremendous velocities, close to the velocity of light. The idea of the photon engine was proposed by the German scientist E. Sänger, but the creation of such an engine depends on the solution of a number of very complicated problems.

Quite recently Sänger proposed a new type of photon engine. In this design the source of photons is a plasma with a temperature of 150,000°C, obtained by means of atomic energy and compressed by strong magnetic fields. However, this project is for the time being only an interesting idea. Its practical realization is associated with a number of serious difficulties. For example, to obtain a parallel jet of plasma (its normal motion is disorderly) a special reflecting mirror with very low absorption of the incident radiant energy is necessary. Ordinary metallic reflectors are unsuitable as the coefficient of reflection of radiant energy falling on them is small.

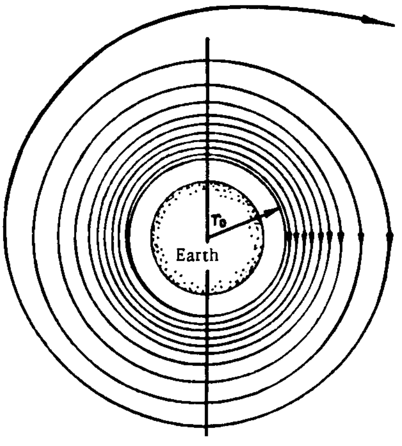

FIGURE 40. Additional acceleration phase of a spaceship with an ion engine

In some cases the second phase of the trajectory flight along a satellite orbit of the departure planet, may be omitted. In this case, at the end of the initial phase the ship is given a velocity which takes it beyond the limits of the earth's sphere of action along a trajectory with respect to the sun.

Flight in the sphere of action of the departure planet is along either an elliptical orbit, whose remotest point is the boundary of the planet's sphere of action, or along a parabolic, or, finally, along a hyperbolic orbit with respect to the planet of departure. Thus, for example, if the departure planet is the earth, it is possible to reach the boundary of its sphere of action by following an elliptical trajectory whose apogee is on the boundary of the sphere of action. Such an orbit results from a starting velocity at the surface of the earth of 11.15 km/sec (disregarding gravitational and atmospheric drag losses).

As the ship leaves the launch planet's sphere of action, its heliocentric motion takes place along an elliptical, parabolic, or hyperbolic orbit with a focus at the center of the sun. The type of orbit depends on the heliocentric velocity, and the orientation of the orbit in space depends on the direction of this velocity. As we want the spaceship to fly rigorously along the calculated orbit, the attainment of the calculated values of the parameters of the heliocentric exit velocity vector is the most important problem of space navigation.

The heliocentric exit velocity vector of the spaceship, in the case of starting from the earth, is equal to the vector sum of the orbital velocity V_E of the earth and the geocentric exit velocity V'_{exit} (Figure 41).

If the heliocentric exit velocity is lower than the escape velocity for the given distance of the ship from the sun, the ship will move along a heliocentric elliptical orbit. When these velocities are equal, the ship will move along a parabolic orbit, and if the exit velocity exceeds the escape velocity, along a hyperbolic orbit.

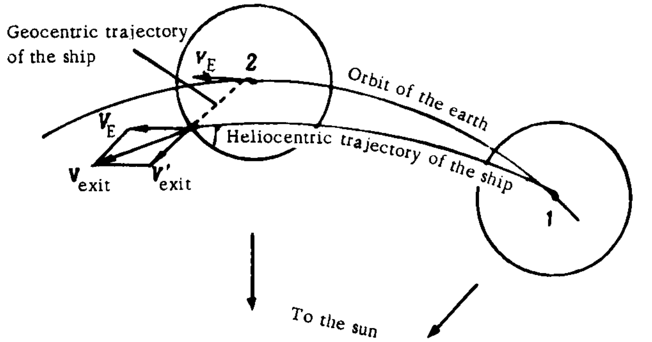

FIGURE 41. Heliocentric exit velocity of a spaceship starting from the earth

1 and 2- the positions of the earth at the moment of launching and at the moment of its leaving the sphere of action of the earth.

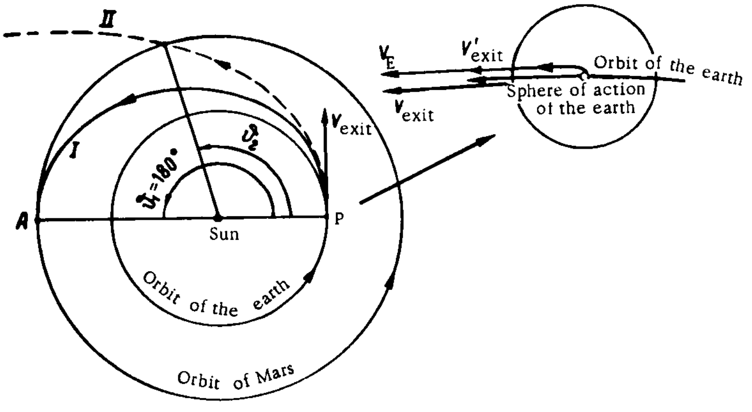

FIGURE 42. Two types of heliocentric elliptical orbits in flight to Mars

I- minimum energy semi-elliptic orbit; II- nonminimum energy orbit, resulting in a shorter flight duration.

Parabolic and hyperbolic orbits require higher exit velocities than elliptical orbits, and therefore from the energy viewpoint the last is most economical. Spaceships using chemical fuels will fly along elliptical orbits. However, flights along elliptical orbits are longer; their duration even when flying to comparatively close planets may be from several months to several years. In the future, with the appearance of powerful atomic and ionic engines, spaceships will fly along shorter "roads" — along parabolic and hyperbolic orbits.

Suppose it is necessary to fly to a superior planet, for example to Mars, in a ship using a chemical-fuel engine. Such a ship can achieve "heliocentric flight" only along an elliptical orbit. The ship is launched so that on the boundary of the earth's sphere of action, the direction of the geocentric exit velocity coincides with the orbital velocity of the earth, as a result of which the heliocentric exit velocity is equal to their arithmetic sum. In principle, two such solar elliptical orbits are apparently possible (Figure 42). The perihelions of the orbits of the first and second types are situated on the orbit of the earth. The aphelion is on the orbit of Mars for the first type of ellipse, and beyond it for the second type of ellipse. It can be easily seen that as far as flight duration is concerned, the second type is more advantageous, since the angular length of the flight is less than 180° (for the first type of orbit it is always equal to 180°). However, we already know that in order to move along a more elongated elliptical orbit (the second in our case) the heliocentric exit velocity should be higher (Table 12).

TABLE 12. Required velocities at the end of the initial phase at H = 200 km from the earth for various aphelion distances of the orbit from the sun

Required velocity at the end of the initial phase, km/sec	Distance from the sun to the aphelion of the orbit, million km
11.015	168.9
11.515	247.7
12.015	314.1
13.015	480.1
14.015	760.3
15.015	1400.0
16.015	4618.0

Any number of orbits of the second type is possible, but although they have shorter flight durations, they all require larger energy expenditures than the unique orbit of the first type. This is why orbits of the first type are called minimum-energy ellipses. Sometimes such orbits are also called semi-elliptical.

Minimum energy orbits have another important disadvantage. As is known, in shooting a rifle, the longer the range the less the accuracy. Something similar occurs in space travel. As the range of a spaceship is increased, the precision of its arrival at the destination decreases. Conversely, if a given accuracy at destination is required, the vehicle must be launched with greater precision for longer ranges. This is particularly important for unguided space probes.

Another constraint exists in the case of minimum energy orbits. As shown in Figure 42, the earth at the moment of launch, and the destination planet, at the moment of expected arrival, must lie on a straight line passing through the center of the sun. In other words, they must lie on the line of apsides of the heliocentric orbit of the spaceship. This situation only occurs at certain times.

Favorable moments for a minimum energy flight to Mars occur approximately every 2.14 years. In 1962, this was from the end of October to the first part of November.

Orbits of the second type for flight to Mars are possible for periods of approximately one year with a subsequent interruption of 13 months.

Let us now consider possible heliocentric elliptical orbits for flight to inner planets, for example to Venus, or back from Mars to the earth. The direction of the ship's velocity vector from the boundary of the departure planet's sphere of action should be opposite to the orbital velocity vector of the departure planet. The heliocentric velocity of the ship is therefore equal to the difference between these velocities and the orbit of the ship will lie inside the orbit of the departure planet. The two orbits are tangential at the aphelion of the ship's orbit. Only along such orbits is it possible to fly to inner planets. Interestingly, such flight requires larger energy expenditures than flight to superior planets. The distance of the perihelion of such orbits depends on the velocity of the spaceship at the end of the initial phase (Table 13).

TABLE 13. Required velocity at the end of the initial phase and distance of the perihelion of the orbit from the sun (height of initial orbit 200 km from the earth)

Required velocity at the end of the initial phase, km/sec	Distance of the perihelion from the sun, million km
11.015	132.8
11.515	95.6
12.015	80.3
13.015	61.8
14.015	49.8
15.015	40.9
16.015	33.9
16.7	30.0

Tables 12 and 13 show that if the velocity at the end of the initial phase exceeds the parabolic velocity by 2 km/sec, the "outbound" ship reaches a distance of 480.1 million km from the sun (330.9 million km from the orbit of the earth). In the second case, when flying to inner planets, the ship approaches to within 61.8 million km of the sun, (only 87.7 million km from the orbit of the earth).

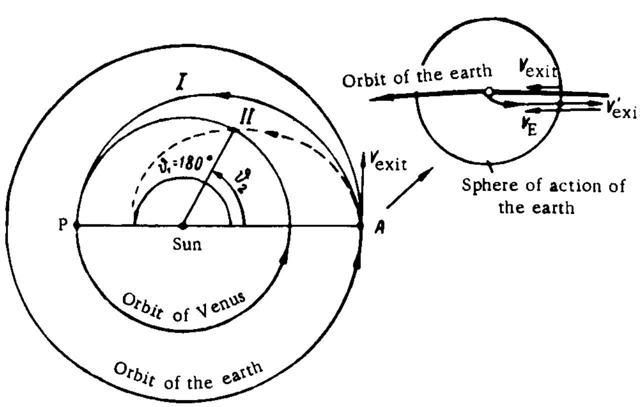

FIGURE 43. Two types of heliocentric elliptical orbits for flight to Venus

I- minimum-energy semi-elliptical orbit; II- nonminimum energy orbit, resulting in a shorter flight duration.

Two types of elliptical orbits are possible when flying to inner planets, too (Figure 43). The orbit of the second type intersects the orbit of the destination planet, while the orbit of the first type is tangent to the orbit of the destination planet. The contact point is at the perihelion of the spaceship's orbit. The tangent orbit requires less energy expenditure since it is a minimum-energy ellipse. But, as in the case of flight to outer planets, minimum-energy flight is more prolonged than flights along orbits of the second type, and requires higher navigational accuracy. All this should be taken into account in planning flights both of manned spaceships and of automatic interplanetary stations with scientific equipment. The longer duration also increases the probability of encountering meteoric bodies which are capable of injuring the spacecraft. Of course, other factors should also be taken into account, for example, the scientific purposes of the launching of the vehicle.

The heliocentric velocity of the spaceship along an elliptical orbit with respect to the sun is given for any point by the already-familiar formula

$$V^2 = R_\odot^2 g_0 \left(\frac{2}{r} - \frac{1}{a}\right) = fM_\odot \left(\frac{2}{r} - \frac{1}{a}\right),$$

where $R_\odot$ is the radius of the sun; g_0 is the gravitational acceleration on the surface of the sun; $M_\odot$ is the mass of the sun; a is the semimajor axis of the ship's orbit; r is the distance from the given point of the orbit to the center of the sun.

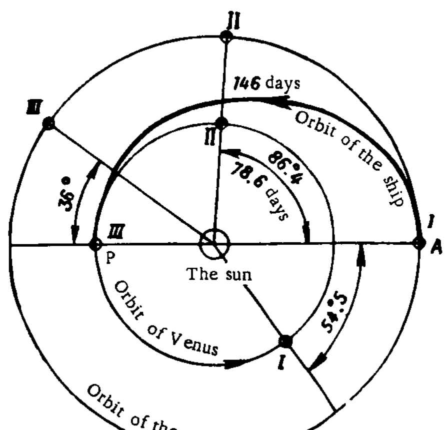

FIGURE 44. Positions of the earth and Venus for a space flight to Venus along a semi-elliptical orbit (Hohman's calculated trajectory)

I and I - the positions of the earth and Venus at the starting moment of the ship; II and II - Venus' conjunction; III and III - the positions of the earth and Venus at the moment of the ship's arrival at Venus.

As an example let us consider the flight of the first automatic interplanetary station (AIS) to Venus, launched in the Soviet Union on 12 February,

1961. Its heliocentric flight was close to a semi-elliptical orbit, corresponding to a minimum-energy ellipse (Figures 44 and 45).

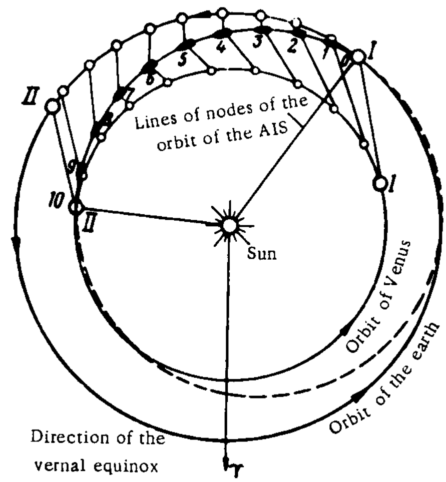

FIGURE 45. Orbit of the first automatic interplanetary station (AIS) with respect to the sun

I and I- positions of the earth and Venus at the starting moment of the AIS; II and II- the moment the AIS approaches Venus; 1, 2, ..., 10 - the successive positions of the AIS with respect to the earth and Venus.

First the AIS was introduced into an approximately circular parking orbit with a perigee of 230 km, an apogee of 287 km, and an inclination of 65°. At a precalculated point of the orbit the AIS was accelerated to a velocity 0.661 km/sec higher than the second cosmic velocity.

Inside the earth's sphere of action the AIS moved along an orbit close to a hyperbola. It reached the boundary of the sphere of action on 14 February at 2300 hours Moscow time. Its velocity with respect to the earth there was about 4 km/sec. Its velocity with respect to the sun, equal to the vector sum of the earth's orbital velocity and the spacecraft's velocity with respect to the earth at the boundary of the earth's sphere of action, was 27.7 km/sec. We recall that the mean orbital velocity of the earth with respect to the sun is 29.8 km/sec.

The AIS then proceeded along an elliptical orbit with one of the foci at the center of the sun. The distance of the orbit's aphelion from the sun was 151 million km, and the distance of the perihelion, 106 million km. The inclination of the orbit to the plane of the ecliptic was 0.5°. The orbital planes of the earth, Venus, and the automatic interplanetary station are slightly inclined with respect to each other. In Figure 45 the orbit of the AIS is shown projected on the plane of the earth's orbit. The simultaneous positions of the earth and the AIS, and of the AIS and Venus are connected by straight lines. As seen from the figure, at the start of the flight the AIS lagged behind the earth (positions 2 and 3). Not long before the vernal equinox, 21 March, the earth, the AIS, and the sun were all situated approximately on a straight line (position 4), and then the AIS in its angular motion around the sun overtook the earth (positions 5, 6, and subsequent ones).

The AIS steadily drew away from the earth and at the moment of approach to Venus was 70 million km distant. Measurements of the orbit indicated that the AIS passed at a distance of 100,000 km from Venus. The moment of closest approach was on 19 to 20 May, 1961. Thus, the flight time of the AIS until the approach to Venus was slightly more than three months. The high accuracy with which the AIS was launched into orbit is an outstanding achievement of Soviet astronautics.

Unfortunately, it was not possible to complete this highly interesting first experimental flight from the earth to another planet, due to the interruption of radio communication with the AIS. This launching, and that of the AIS "Mars I", indicate the success of powerful Soviet rocket systems and highly accurate guidance systems in pioneering flights to other planets.

We shall conclude the characterization of the heliocentric flight section of a spaceship by considering the flight duration. We note at once that the duration of this section of the trajectory is considerably longer than the duration of its other phases. The total flight duration to other planets will therefore be determined mainly by the flight time along the heliocentric orbit.

For flight along a semi-elliptical orbit, the flight duration will be equal to half the period of revolution:

$$t=\frac{P}{2}=\sqrt{\frac{\pi^2 a^3}{fM_{\odot}}},$$

where a is the semimajor axis of the orbit.

This flight time even to the nearest planets is comparatively long (Table 14). Therefore, in a number of cases it is necessary to plan the flights along orbits of shorter duration, although they require larger energy expenditures. Calculations show that in this case the flight duration can be shortened by a factor of 2 or more (Table 15). However, specialists think that the use of such orbits is connected with the problem of creating new powerful engines for spaceships running on nonchemical fuels.

TABLE 14. Velocity at the end of the initial phase at H = 200 km and the flight duration along a semi-elliptical orbit until arrival at a planet of the solar system

Planet	Velocity at end of initial phase, km/sec	Flight duration until arrival at the planet, years
Mercury	13.31	0.29
Venus	11.25	0.40
Mars	11.35	0.71
Jupiter	14.05	2.72
Saturn	15.03	6.04
Uranus	15.73	16.0
Neptune	16.00	30.6

TABLE 15. Velocity at the point of entering an elliptical orbit with respect to the sun and the flight duration for various angular distances

Velocity at the moment of entering an elliptical orbit with respect to the sun, km/sec	Angular distance, degrees	Flight duration, months
	To Venus	
27.28	180	4.87
26.28	110	3.33
25.28	89	2.83
24.28	76	2.52
23.28	66	2.33
22.28	59	2.16
	To Mars	
32.71	180	8.63
33.71	124	5.25
34.71	108	4.32
35.71	97	3.77
36.71	90	3.40
37.71	85	3.10

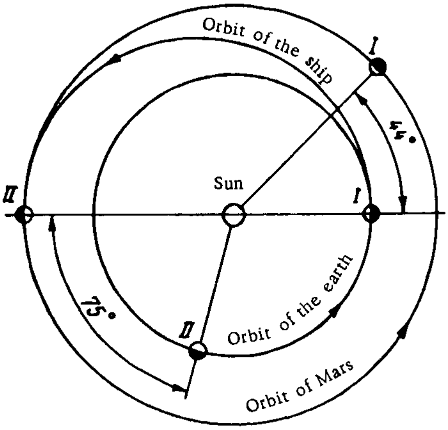

FIGURE 46. Positions of the earth and Mars for flight to Mars along a semi-elliptical orbit

I and I- positions of the earth and Mars at the starting moment of the ship; II and II- at the moment of arrival on Mars.

For flight from the earth to another planet, the moment of launching of the spaceship should correspond to a favorable position of the planet with respect to the earth. For example, positions of Mars favorable for flights, as shown in Figures 46 and 47, occur approximately once every 2.14 years, and favorable positions of Venus, once every 1.57 years. It is also necessary to wait for a favorable position of the planets when returning to the earth, which considerably increases the total duration of round-trip interplanetary flight.

According to one set of calculations, after reaching circular orbit around Mars, or after landing on it, the waiting time for a return start to earth is 452 days. The total flight duration from the earth to Mars and back again amounts in this case to 970 days, or 2 years 8 months.

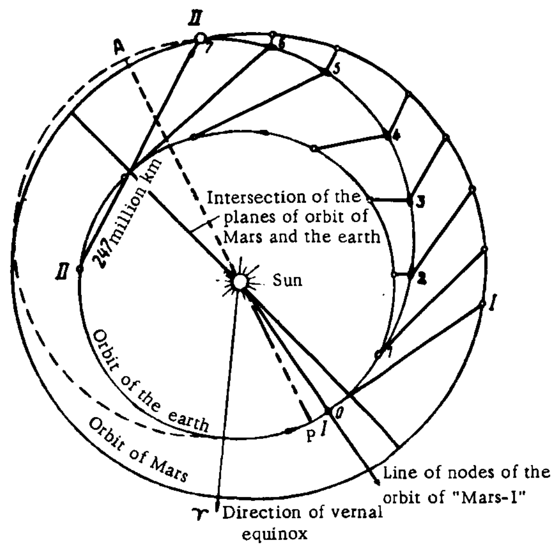

FIGURE 47. Orbits of Mars and of the AIS "Mars I" projected on to the plane of the earth's orbit. The simultaneous positions of the earth and Mars (I and I- at the moment of launch, and II and II- at the moment of approach of the AIS to Mars) and the positions of the station itself between the moments of launch and approach are joined by lines. Due to the noncoplanarity of the orbits of the earth and Mars, the station, as it approaches Mars, is put into an orbit with an inclination of 2°37', approximately that of the orbit of Mars

Let us now consider the section of the flight of a spaceship in the sphere of action of the arrival planet. The motion of the spaceship in the sphere of action of the arrival planet is determined by its heliocentric velocity on the boundary of this sphere and by the orbital velocity of the planet itself.

Let the ship have a heliocentric velocity V_{ent} on the boundary of the sphere of action of the arrival planet (Figure 48), and let the heliocentric orbital velocity of the arrival planet be V_{orb}. To determine the velocity of the ship with respect to the arrival planet it is necessary to subtract the arrival planet's velocity from the ship's entrance heliocentric velocity vector.

If the resultant velocity V'_{ent} is lower than the escape velocity at the boundary of the sphere of action, the further motion of the ship with respect to the arrival planet will be along an elliptical trajectory. If V'_{ent} equals escape velocity, the motion will be along a parabola, and if it exceeds escape velocity, the path will be a hyperbola. Thus the magnitude of the entrance velocity V'_{ent} determines the ship's orbit within the sphere of action of the destination planet.

In principle, the spaceship can approach the boundary of the sphere of action of a celestial body with a heliocentric velocity close to zero. It is obvious that in this case the magnitude of the velocity V'_{ent} will be close to the orbital velocity of the planet, but oriented in the opposite direction. The minimum value of V'_{ent} occurs when the spaceship is moving opposite to the planet's orbit. In this case, V'_{ent} equals the algebraic sum $V_{orb} - V_{ent}$. But this will usually require an additional maneuver of the ship, so in most cases V'_{ent} will be equal to or greater than V_{orb}. The following ratio serves as an index of the form of the orbit inside the sphere of action of the destination planet

$$\xi = \frac{V_{orb}}{V_{c_2\,(sp.ac.)}},$$

where $V_{c_2(sp.ac.)}$ is the escape velocity on the boundary of the sphere of action of the destination planet.

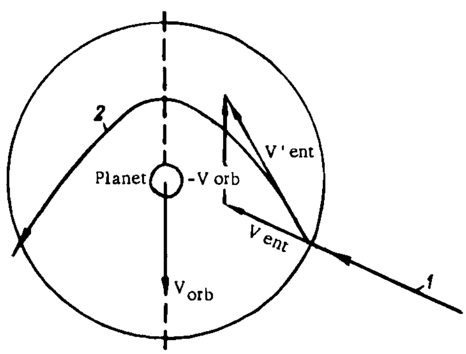

FIGURE 48. Determination of the velocity of a spaceship on the boundary of the sphere of action of the arrival planet

1-heliocentric trajectory; 2- trajectory in the sphere of action of the arrival planet.

Only for $\xi < 1$ is elliptical motion of the ship with respect to the destination planet possible.

Thus, for example, in the case of Venus the orbital velocity is 34.99 km and the zero-height escape velocity is 10.2 km/sec. On the boundary of the sphere of action the escape velocity will be lower and the index ξ will be larger than 1. Consequently, spaceships which fall into the sphere of action of Venus move within it along hyperbolic orbits. Such was the motion of the first Soviet automatic interplanetary station to Venus within its sphere of action.

The same can be said about the earth, whose orbital velocity is 29.8 km/sec, and whose escape velocity at the surface is 11.2 km/sec. Ships returning to the earth from outer space will move along hyperbolic orbits inside its sphere of action.

It is obvious that the value of ξ will decrease for planets situated further and further from the sun, resulting from a decrease in their orbital velocities. However, a definite role is played in this matter by other factors too, and a qualitative estimate of the extent of variation of the index ξ for outer planets, particularly for the giant planets Jupiter, Uranus, and Saturn, is difficult.

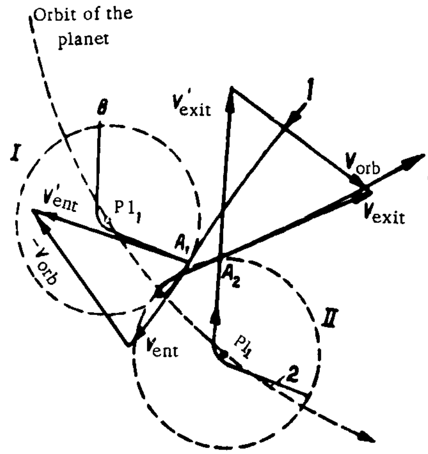

FIGURE 49. Entrance and exit velocities of a spaceship on the boundary of the sphere of action of the destination planet. Perturbation maneuver:

1- heliocentric trajectory; 2- trajectory in the sphere of action of the planet; I and II- positions of the planet and its sphere of action at the entrance and exit moments of the ship.

The ship's encounter with the destination planet is independent of the form of the orbit, if the latter passes at a distance from the planet's center, which is less than its radius. It is obvious that if the distance is larger than the radius of the destination planet, the ship does not meet the planet. Along parabolic and hyperbolic orbits the ship will emerge from the planet's sphere of action and will continue moving along an orbit with respect to the sun.

It is interesting to note that in the case of parabolic or hyperbolic motion in the sphere of action of a planet, the velocities V'_{ent} at the point of entrance to the sphere of action and V'_{exit} at the point of exit from the sphere of action are equal in magnitude, but have different directions (Figure 49). Let the point A_1 be the point of entrance of the ship into the sphere of action of the planet. The heliocentric velocity V_{ent} of the ship at this point is determined by the tangent to the heliocentric trajectory. The "planetocentric" velocity V'_{ent} (the velocity with respect to the planet) is equal to the vector sum of the entrance heliocentric velocity and the orbital velocity of the planet, taken with an opposite sign. The velocity V'_{ent} determines the motion of the ship with respect to the destination planet along the hyperbolic orbit A_1B.

When the ship approaches point B the planet is at the point Pl_2 and point B coincides with the point A_2, the point of exit of the ship from the sphere of action of the planet. At this point the exit "planetocentric" velocity V'_{exit} is equal to the entrance "planetocentric" velocity V'_{ent}, but their directions are different. This is due to the fact that the points A_1 and B of the "planet-ocentric" trajectory are symmetric with respect to the planet.

We may now examine the exit heliocentric velocity of the ship. For this purpose the vectors V'_{exit} and V_{orb} at the point of exit must be added.

As can be seen from Figure 49, the entrance V_{ent} and exit V_{exit} heliocentric velocities of the spaceship are different both in magnitude and in direction. This can be used in space flight for maneuvering the spaceship without any fuel expenditure.

This type of spaceship maneuver, i.e., flying along a parabolic or hyperbolic orbit in the sphere of action of some celestial body, is called a perturbation maneuver.

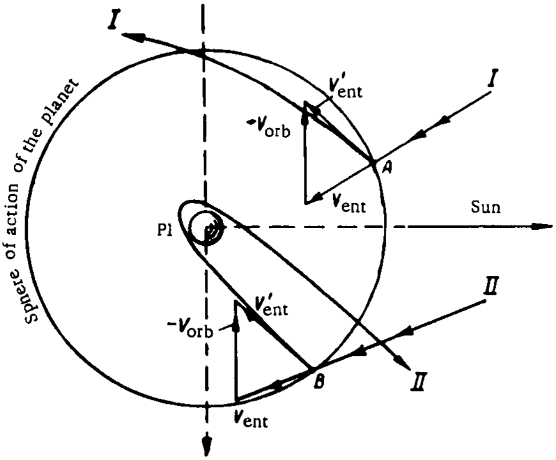

FIGURE 50. Direct I and retrograde II motion of a spaceship inside the sphere of action of the destination planet

We draw the reader's attention to another important fact connected with the motion of a spaceship in the sphere of action of the destination planet. Depending on the position of the entrance point of the spaceship to the sphere of action of the destination planet, its motion with respect to the destination planet can be direct as well as retrograde (Figure 50).

This should be taken into account in planning the flight, particularly when it is necessary to land on the surface of a planet having a high rotational velocity about its axis.

Parabolic and hyperbolic trajectories for flying around the destination planet will obviously be used for automatic interplanetary stations in order to study the nearest planets of the solar system. For spaceships launched with the purpose of a prolonged study of the planet, and in the future also for landing on it, these trajectories are inadequate. The best orbits inside the sphere of action are elliptical with direct motion, and therefore the problem of transfer from parabolic and hyperbolic to elliptical orbits arises.

This transfer can be accomplished only by either reducing the ship's velocity with respect to the destination planet, or by reducing and simultaneously changing the orientation of the ship's velocity vector (Figure 51, points A and B).

Thus, a spaceship can be transferred to an elliptical and even to a circular orbit with respect to the destination planet, either for a prolonged study, or for subsequent descent to the surface. Direct descent to the surface of the destination planet from parabolic and hyperbolic orbits is also possible.

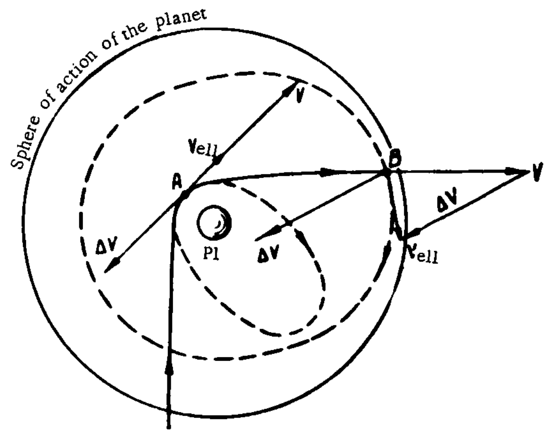

FIGURE 51. Possible variations in transfer to an elliptical orbit with respect to the destination planet (ΔV- the decrease in the velocity by application of reverse thrust)

Finally, the last phase of the trajectory of an interplanetary flight is descent to the surface of the destination planet. This is the most complicated problem of interplanetary flight. For example, at the time of landing, the spacecraft with the dogs Belka and Strelka had a velocity of 0.06 to 0.8 km/sec, whereas in orbit at a height of 300 km, its velocity was about 8 km/sec. Consequently, on the descent section, the flight velocity has to be reduced by a factor of one thousand or more /22/. This example indicates the difficulties which the scientists face in solving this problem.

The problem of a safe landing on the surface of the destination planet can, in principle, be solved in two ways: by means of rocket engines and by using aerodynamic forces, or by a combination of the two. However, it should be borne in mind that aerodynamic forces can be used only when the destination planet has a sufficiently dense atmosphere (earth, Venus, and others).

The first method of landing requires large energy expenditures, first, to reduce the velocity of the ship, and second, to compensate for the gravitational force on the descent phase. The relative energy expenditures for descent by the first method can be determined from the magnitude of the characteristic velocity, equal to the sum of the ship's velocity with respect to the destination planet at the beginning of the descent phase, and the compensation velocity for the gravitational force (Table 16).

The high characteristic velocity of the descent phase is one of the reasons for the small payloads of ships with engines using existing chemical fuels. Thus, for example, when using propellant with an exhaust velocity of 4000 m/sec, the ratio of the payload to the weight of the whole ship amounts, in the case of landing on the earth and on Venus, to less than 10%, on the moon, from 30 to 60%, and on Mars, from 15 to 30%. In the case of landing on Jupiter and Saturn, the weight of the payload will be practically zero /2/.

TABLE 16. Examples of characteristc velocities for landing on the surface of the earth, the moon, and other planets

Planet	Characteristic landing velocity, km/sec	
	from a parabolic orbit	from a circular orbit at a height of 1000 km
Earth	13.0	8.1
Venus........	12.0	7.4
Mars	5.8	3.4
Jupiter	68.5	46.0
Saturn........	41.0	27.5
The moon	2.5	1.5

If there is an atmosphere at the destination planet it can be used for descent by the second method. Two types of descent with aerodynamic braking are possible: descent along a ballistic trajectory and descent along a gliding trajectory. In the first case the ship without lifting surfaces moves with a zero angle of attack.

The disadvantages of this method are the tremendous braking decelerations and very high final velocities of the spaceship. For safe landing in this case, braking parachutes have to be used. In addition, such a descent is accompanied by the release of a huge amount of energy. The kinetic energy possessed by the spaceship is transformed into heat. As a result of this, the temperature of the air-stream flowing around the ship rises and the body of the ship is intensely heated. Thus, according to one calculation, for each kilogram of weight of a ship descending to the surface of the earth, about 5500 kcal are released. Even for a ship of comparatively small weight, about 450 kg, a huge amount of heat is released — as much as 2.5 million kcal. This energy is sufficient to destroy a ship of any normal construction /22/.

One of the effective ways of dissipating the heat received by the spaceship is to use a protective layer for the body. This layer, also called an ablative shield, consists of a material which absorbs a great amount of heat in melting and evaporating. One such material is beryllium oxide. For thermal protection of a ship weighing 450 kg descending in this way to the earth, it is necessary to evaporate approximately 210 kg of beryllium oxide /22/.

Another method for combating the thermal danger on the descent phase is to give the spaceship a special aerodynamic form. It is assumed that for this purpose the best configuration is a blunt nose. With this form, compression and intensive heating of the air essentially occurs in front of the ship, and the forward section of the ship is heated to a comparatively low temperature (1100 to 1300°). The rest of the ship's surface is outside the zone of intensive air heating and therefore remains practically unheated /22/.

What is the possible temperature rise in the zone of air compression in front of a descending ship? In foreign literature the following data are given for a spaceship in the form of a disk, entering the earth's atmosphere at an angle of 45° to the horizon with a velocity of about 11.2 km/sec: the pressure on the frontal surface of the ship facing the stream exceeds the pressure of the surrounding air by a factor of 1085, and the temperature may reach 50,000°.

The higher the relative entrance velocity into the atmosphere, the higher the rate of heat production. Trajectories of spaceships therefore should be planned so that the entrance of the ship into the dense layers of the atmosphere will be in the direction of the diurnal rotation of the destination planet with its surrounding atmosphere.

Gliding descent at small negative inclination angles of the velocity vector to the horizon makes it possible to reduce gradually the velocity of the ship under low braking deceleration and comparatively little heating of the ship's body. In addition, a gliding descent makes possible range and direction maneuverability for landing at a fixed point on the surface of the destination planet. The duration of a gliding landing and correspondingly the distance covered naturally exceed those of descent along a ballistic trajectory.

It is possible to begin a gliding descent from those heights where the air density provides sufficient lifting force. If the lifting force of the wings is small, rocket engines giving vertical thrust must be started in order to avoid an excessively steep descent and the resulting large braking loads and heating of the ship. In addition, this method of descent involves tremendous technical difficulties. It is necessary first of all to ensure stability and control of the ship within a very wide velocity range from the huge cosmic to low landing velocities, which clearly requires supporting surfaces of variable area. The external shell of the ship must also withstand heating to between 1500 and 2000°.

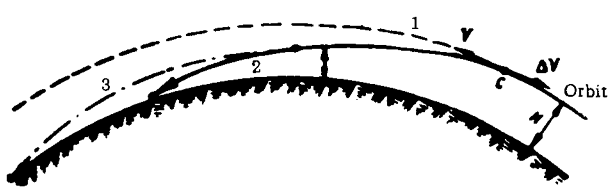

FIGURE 52. Descent trajectories of spaceships:

1- descent phase outside the dense layers of the atmosphere; 2- atmospheric phase of the trajectory in the case of ballistic descent; 3- atmospheric phase of the trajectory in the case of gliding descent.

The possible descent trajectories of a spaceship to the surface of the earth from a circular orbit depend on the additional velocity ΔV which is given to the ship in the direction opposite to its orbital motion. The angle of entrance into the atmosphere also depends on the magnitude of this velocity (Figure 52). The higher the additional velocity, the larger the entrance angle. For a velocity of 0.2 to 0.3 km/sec, the entrance angle is a few degrees. Subsequent motion follows either a ballistic or a gliding trajectory.

On 19 August, 1960, the second Soviet spaceship with living beings on board was launched into orbit as an earth satellite. The command for landing in a designated region was given on the ship's 18th revolution. From this moment until an altitude of 7 km above the surface of the earth was reached, the ship traversed a distance of 11,000 km. The maximum deceleration under braking reached 10 g. At a height of 7 to 8 km the container with the animals was catapulted from the ship's cabin. The container landed at a velocity of between 6 and 8 m/sec, and the ship's cabin, at 10 m/sec. The deviation of the actual from the designated landing point amounted to less than 10 km, a testimony to the successful solution of the extremely complicated problem of descent to the surface of the earth. This experience will be of assistance in solving the problem of descent to the surface of other celestial bodies.

It should be noted that all our considerations were based on the assumption that the planes of the orbits of the departure and destination planets coincide. This is however not completely true, as can easily be seen by comparing the inclinations of the orbits of the planets of the solar system (see Figure 11).

The fact that the planes of the orbits of the departure and destination planets do not coincide, or are noncoplanar, creates additional difficulties in the execution of interplanetary flights. It requires maneuvering the ships in flight, and consequently, the solution of a series of additional navigational problems.

§7. Flight Trajectories to the Moon

In view of the comparatively short distance from the earth to the moon, the first spaceships flying to other celestial bodies will go to the moon. In fact, the minimum distance from the earth to Mars is 54 million km, whereas the mean distance from the earth to the moon is only 384,400 km, i.e., over 135 times shorter than the distance to Mars. It should also be noted that the trajectories of spaceships do not follow the shortest distances between celestial bodies.

In principle, various types of flight to the moon are possible: direct flight to the moon, flight around the moon with return to the earth, orbital flight around the moon, and flight to the moon with entrance into the orbit of an earth satellite.

For each of these types both direct starts from the earth and starts from an orbit around the earth are possible. The initial phase and orbital flight around the earth have already been described in the previous chapter. Therefore, we will not deal with them here, and will refer for simplicity to a ship starting directly from the earth.

The first type is a direct flight to the moon. For flying from the earth to the moon orbits of any form can in principle be used. However, as in the case of interplanetary flights, orbits which require minimum energy expenditure are of particular interest. These earth-satellite elliptical orbits whose apogees coincide with the orbit of the moon (Figure 53a).

When starting from point A of the terrestrial surface, the launch point is the perigee of the orbit. When starting from point B, the perigee is situated inside the earth ($P/_2$), and when starting from point C, the elliptical orbit is transformed into a rectilinear one. The perigee of this orbit is situated at the center of the earth. These three orbits are not equivalent from the energy point of view. The smaller the semimajor axis of the orbit, the lower the energy expenditure in moving along it. Consequently, the most advantageous is the third orbit. However, as is shown by calculations, the difference in energy expenditure for the first and third orbits is very small, and can be practically neglected. In fact, the required initial velocity for flight along the third orbit is approximately 11.09 km/sec, and along the first orbit, 1 to 2 m/sec more, which amounts only to hundredths of a percent.

Thus, in spite of the fact that for each starting point on the earth there is an elliptical minimum-energy orbit, they all require practically the same initial velocity — about 11.09 km/sec. This is approximately 1% lower than the second cosmic velocity for the surface of the earth. There are also other orbits along which lunar flight is possible (Figure 53b): an elliptical orbit for landing on the opposite side of the moon, and parabolic and hyperbolic orbits.

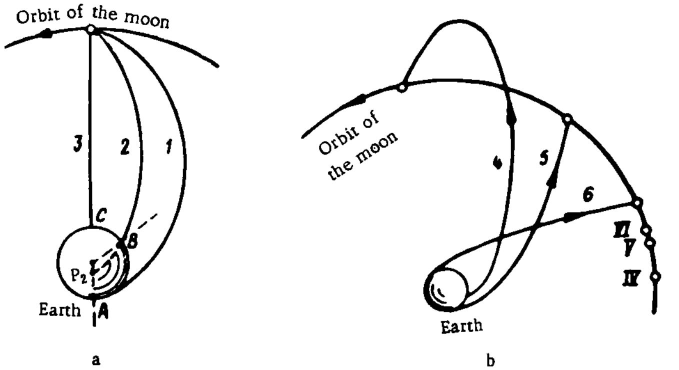

FIGURE 53. Orbits for flights to the moon in the case of a ship starting directly from the surface of the earth

a- minimum-energy elliptical orbits; b- other orbital types; 1 and 2- elliptical orbits starting from the points A and B; 3- radial trajectory; 4- elliptical orbit for landing on the opposite side of the moon; 5- parabolic orbit; 6- hyperbolic orbit (IV, V, and VI – positions of the moon at the moment of launching for flight along orbits 4, 5, and 6, respectively).

It is perfectly clear that hyperbolic and parabolic orbits give shorter flight periods to the moon. What is the possible duration of such a flight?

Along minimum-energy orbits it is practically the same, i.e., for the mean distance from the earth to the moon, approximately 5 days. When flying along orbits requiring higher initial velocities, the flight duration sharply decreases. Thus, for an initial velocity which is higher than the minimum by only 0.05 km/sec, the flight duration is reduced by a factor of two. For an initial velocity equal to the second cosmic velocity, which corresponds to a further increase in the velocity by approximately

0.05 km/sec, the flight duration to the moon is two days, and for an initial velocity exceeding the escape velocity by 0.5 km/sec only one day. However, further increases in the velocity result in a proportionately smaller and smaller decrease in the flight periods.

When flying along any orbit, as the lunar ship gets farther from the earth, its geocentric velocity continues to decrease. The instantaneous value of the velocity can be determined by the formulas given above (§3, this chapter).

Let us now consider the motion of a spaceship in the sphere of action of the moon. The entrance selenocentric velocity, as we now already know, is equal to the vector difference of the geocentric velocity of the ship on the boundary of the moon's sphere of action and of the orbital velocity of the moon.

The orbital velocity of the moon is approximately 1.02 km/sec, and the escape velocity on the boundary of its sphere of action is $V_{c_2(\mathrm{sp.ac.})} = 0.385$ km/sec. Therefore the index of the orbit form inside the moon's action is $\xi > 1$. This indicates that inside the lunar sphere of action characteristic orbits are hyperbolic. Consider the case where the ship has a geocentric velocity equal to zero on the boundary of the sphere of action. Its velocity with respect to the moon, the selenocentric velocity, is therefore equal to the velocity of the moon in its orbit around the earth, i. e., 1.02 km/sec, which considerably exceeds the escape velocity on the boundary of the lunar sphere of action.

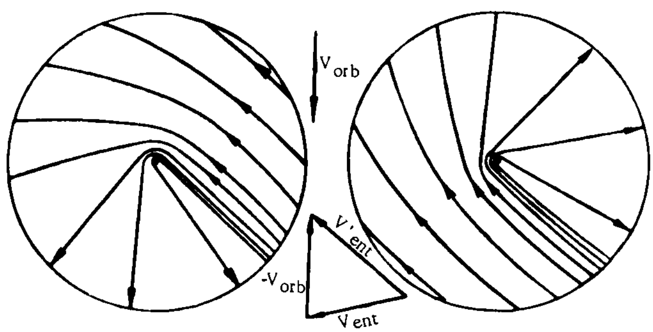

FIGURE 54. Hyperbolic trajectories for different entrance points of a spaceship into the lunar sphere of action

Thus, at a distance of 66,000 km from the moon the ship enters its sphere of action, and its further motion with respect to the moon will be along a hyperbolic orbit (Figure 54). The form of the orbit depends on the position of the entrance point into the lunar sphere of action.

It is obvious that the ship will meet the lunar surface if its orbit approaches the moon's center at a distance smaller than its radius (1740 km).

All the previous considerations did not provide for a modification of the ship's orbital parameters and for guiding it on the orbital section. The accuracy to which the calculated orbital and basic launch parameters can be maintained is therefore of interest.

We shall give for this purpose some calculated figures. An error in the initial velocity of a lunar ship of only 1 m/sec, i. e., approximately of

0.01%, results in a deviation of the point of the ship's encounter with the moon by 250 km. A deviation of the initial velocity vector from the calculated direction by 1 minute of arc results in a displacement of the point of encounter by 200 km. To be sure of hitting the moon, the error in the initial velocity should not exceed a few meters per second, and the inclination of the initial velocity vector should be within 0.1° /6/.

Launching at the exact calculated moment is necessary because the plane of the trajectory of the lunar ship turns in space together with the earth, and also because the moon in its daily motion moves among the stars at the rate of 13.2° per day. An error of 10 sec in the launching moment results in a displacement of the point of encounter by 200 km.

These data show the extremely stringent requirements on the accuracy of the guidance systems of spaceships in the initial phase and on the organization and preparation of the launching.

The possibility of guiding a spaceship on the orbital flight phase, i.e., on the passive phase of the trajectory, allows us to reduce considerably the requirements on the accuracy of the launch parameters of the rocket. However, correction of the orbit of the spaceship requires an accurate determination of its current coordinates. This is one of the basic problems of space navigation. The possible methods of its solution will be shown in the following chapter.

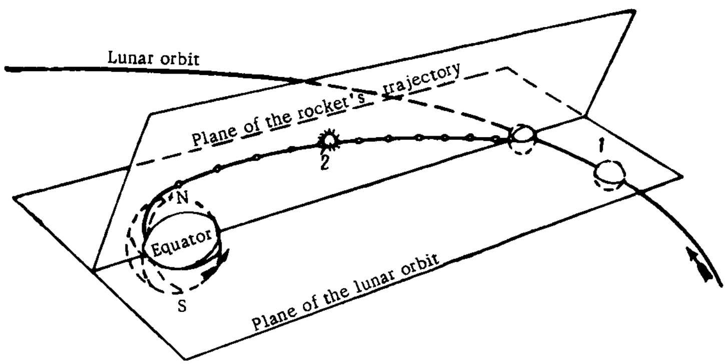

FIGURE 55. Flight trajectory of the second Soviet lunar rocket

1- position of the moon at the launching of the rocket; 2- position of a hypothetical comet on the rocket trajectory.

A brilliant example of flight to the moon is the flight of the Soviet lunar rocket, launched on 12 September, 1959. We recall some of the data of this first flight to the moon. The last stage of the rocket exceeded the second cosmic velocity, and it therefore followed the rocket along a hyperbolic orbit (Figure 54). The velocity of the rocket with respect to the earth on the boundary of the moon's sphere of action was about 2.31 km/sec. Inside the moon's sphere of action, at a distance of 1000 km from its surface, the rocket's velocity with respect to the moon was 2.97 km/sec.

The rocket entered the moon's sphere of action on 13 September, at 16 hrs 40 min, Moscow time, and at 0 hrs 02 min 24 sec on 14 September it reached the lunar surface. The duration of the flight was approximately one and a half days. The rocket container with the instruments landed near the craters Aristullis, Archimedes, and Autolycus. The distance from the landing point to the center of the visible disk of the moon was approximately 800 km. At the moment of impact, the trajectory was inclined at an angle of 60° to the surface of the moon, and the selenocentric velocity was about 3.3 km/sec.

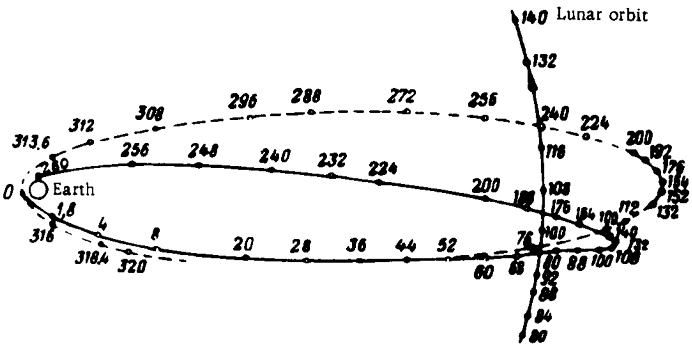

FIGURE 56. Trajectory of a flight around the moon according to K. Erik and G. Gamow (USA) (the figures give the flight time in hours from the launching point; the broken curve gives the orbit of the same spaceship disregarding the moon's influence)

Let us now consider the motion of a spaceship along trajectories taking it around the moon. By this we will understand those which take the ship beyond the orbit of the moon with subsequent return to the earth. Such trajectories (Figures 56 to 59), as well as trajectories leading to a landing on the moon, are of definite scientific interest for investigation of the moon and the space near the moon. These will probably remain the only ones for manned spaceships for a long time, since impacting trajectories require, first of all, "soft landing," and, secondly, take-off from the moon in order to return to the earth. The solution of the latter problem is for the time being distant; but the time will doubtless come when spaceships will start from lunar cosmodromes to the earth and other planets.

The idea of flying around the moon was first proposed by Jules Verne in his novels "From Earth to Moon" and "Around the Moon." His heroes Barbiken, Nikolai, and Ardan, fly around the moon inside an artillery projectile. Now lunar flight no longer seems fantastic, but powerful rocket engines, rather than artillery, make it a possibility. It is obvious that before long Soviet astronauts, like Jules Verne's heroes, will actually make a remarkable journey around the moon.

Now it can be stated that the geometrical form of the moon-orbiting trajectory, particularly where it passes through the sphere of action of the moon, can be most varied with respect to the earth. It was shown in the previous section that the exit velocity with respect to the destination planet is equal to the entrance velocity. This also holds true on the moon. But with respect to the earth, both the magnitude and the direction of the exit velocity may be different. Therefore, the trajectory of a vehicle after leaving the moon's sphere of action will in general differ both in form and position with respect to the earth from the trajectory of flight to the moon. Very small deviations from the calculated point of entrance of the spaceship to the moon's sphere of action, as well as small deviations of the velocity vector may lead to a considerable deviation of the actual return orbit from the predicted one. This situation complicates the accomplishment of a flight around the moon with return to the earth. Such a flight is apparently possible only on a guided ship, which can determine the errors in the motion parameters and correct the actual trajectory.

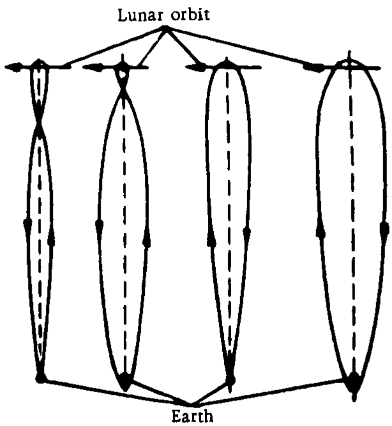

FIGURE 57. Trajectories of symmetrical flight around the moon according to M. S. Lisovskaya (USSR). The minimum distance of the trajectory from the moon is 6378 km

When flying along a symmetrical trajectory around the moon (Figure 57), after leaving the earth's sphere of action the ship will fly identically on the two branches of the trajectory. However, if on returning it does not meet the earth (second and fourth cases), it intersects the outbound branch of the flight trajectory to the moon at some angle. As a result of this, it can no longer fly along the previous trajectory to the moon. Such trajectories are called unclosed.

A peculiarity of the trajectories shown in Figure 58 is the fact that their last section is radial, directed to the center of the earth. This "alignment" of the trajectory is made by the moon. As can be seen, such trajectories are possible both in the case of a close approach to the moon (the upper two trajectories) and in the case of a comparatively distant approach (the lower trajectories). The flight time along round-the-moon trajectories which come near the moon is 5 to 10 days.

Trajectories with a radial return section and symmetric trajectories have the disadvantage that they do not provide for a possibility of safe descent of the spaceship to the surface of the earth. We already mentioned the necessity of a slanting descent at small angles to the horizontal. The problem of the calculation of a trajectory around the moon with entrance into the terrestrial atmosphere at small angles is called the special problem of flight around the moon. Some of the possible trajectories for flight around the moon with slanting return to the atmosphere, obtained by V. A. Egorov, are shown in Figure 59.

A disadvantage of trajectories which come near the moon is that they impose stringent requirements on accuracy in achieving the calculated launching parameters.

Thus, for one of the round-the-moon trajectories with an initial velocity of 10.92739 km/sec, the ship should pass 12,900 km from the center of the moon and return to the earth. If the initial velocity is lower than the calculated value by only 10 m/sec (0.01 km/sec) or if the direction of the initial velocity vector varies by 5° above the horizontal, the ship will either collide with the moon, or will not pass over the center of the hidden side of the moon /19/.

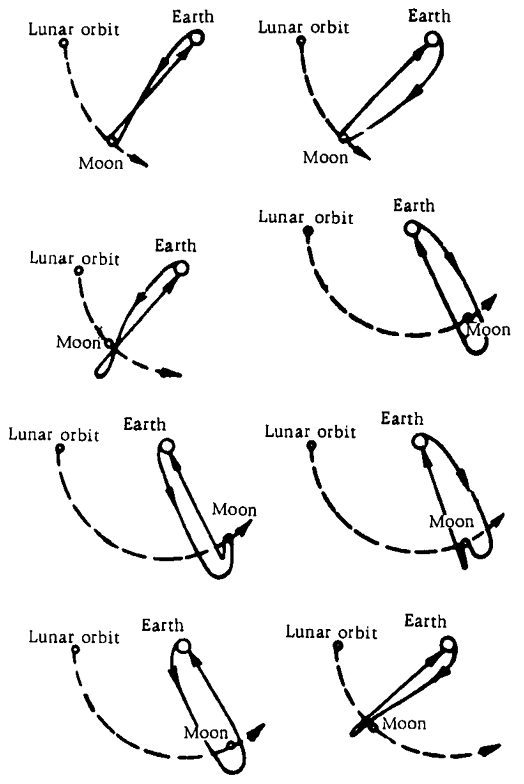

FIGURE 58. Orbital trajectories with radial section in the return flight to the earth, calculated by V.A.Egorov (USSR)

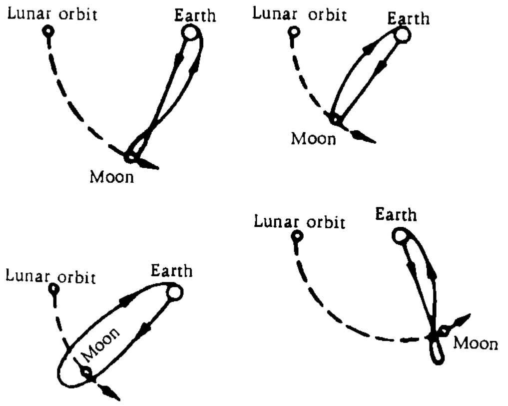

FIGURE 59. Flight around the moon with a slanting re-entry into the earth's atmosphere

As the minimum altitude of the trajectory to the moon increases, the requirements for the accuracy of the launch parameters become less stringent.

Trajectories of periodical flight around the moon are of great scientific interest (Figure 60). The period of revolution of the satellite ship of the earth along the upper orbit is exactly half the revolution period of the moon around the earth. Therefore, after one "idle" revolution without flying around the moon, on the next revolution the ship passes the moon. The ship flies around the moon once every sidereal month.

The revolution period of the satellite-ship along the lower orbit is 2/5 of the revolution period of the moon (2/5 of a sidereal month).

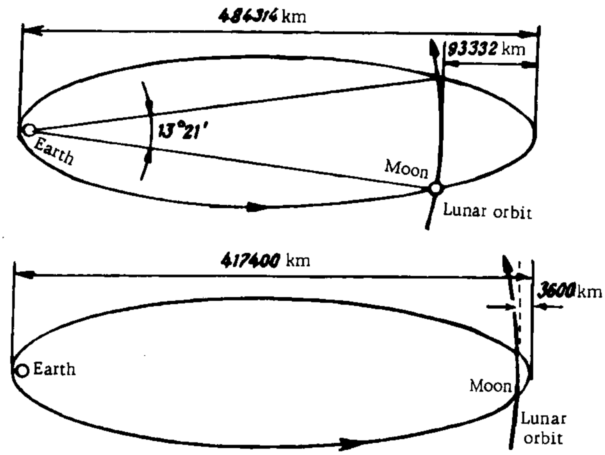

FIGURE 60. Trajectories for periodical flight around the moon with return to the earth, according to A. Shternfel'd (USSR), calculated without taking into account lunar influence

An example of a practical solution of the problem of flight around the moon is the flight of the third Soviet cosmic rocket, launched on 4 October, 1959 (Figures 61, 62, and 63). The objective of the mission was to obtain photographs of the opposite side of the moon and to transmit them to the earth, and therefore a number of special requirements were imposed on the flight trajectory. The distance at the photographing moment could not exceed 60 to 70 thousand km. For normal operation of the camera orientation system during the photographing of the moon, the station and the moon, and the sun had to lie approximately on one straight line. The flight trajectory also had to make it possible for ground receiving points in the USSR to receive the maximum amount of information from the station on the first revolution, preferably at short distances from the earth.

In addition, to carry out a program of scientific investigations, it was desirable to obtain a trajectory along which the automatic station would fly for a sufficiently long time after going around the moon.

As can be seen from Figure 63, the automatic stations passed at a distance of 7900 km from the center of the moon. At the moment of maximum

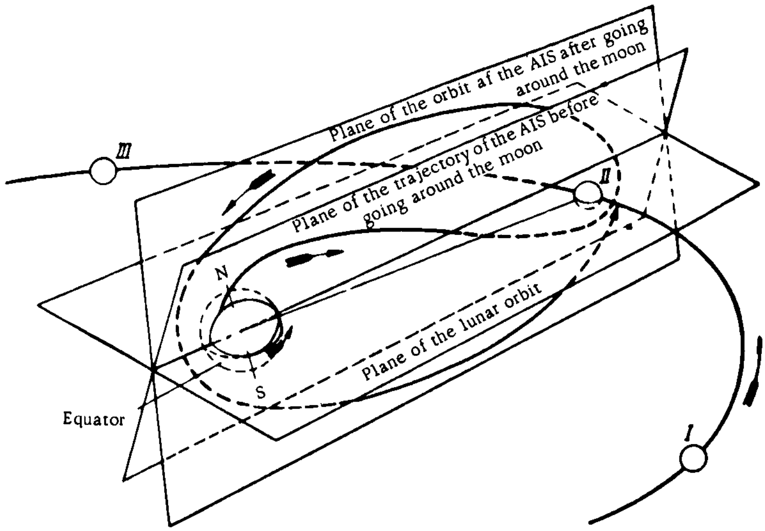

FIGURE 61. Spatial scheme of the flight trajectory of the AIS to the moon

I, II, and III- positions of the moon on its orbit at the moments when the AIS started, approached the moon, and returned to the earth, respectively.

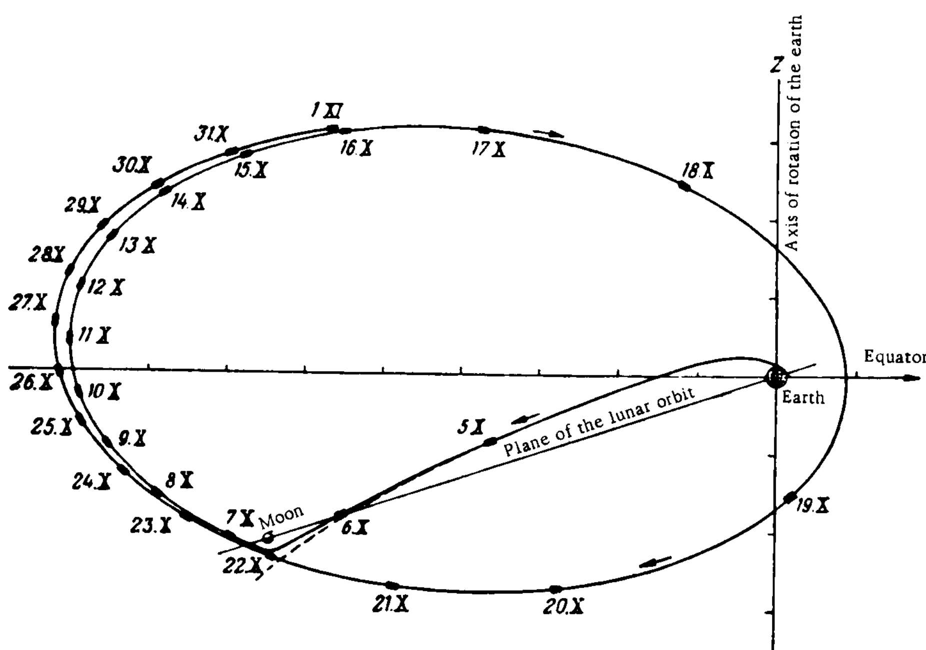

FIGURE 62. Form of the trajectory of the AIS from the direction of the vernal equinox. The positions of the AIS and of the moon on the orbits are given for 0300 hours Moscow time each day

approach, the station was situated south of the moon (Figure 62). Due to the moon's attraction, the station's trajectory was then deflected to the north. The deflection was so large, that the return to the earth was from the northern hemisphere. After flying past the moon, the height of the station above the horizon for observation points in the northern hemisphere increased each day. As another result, the possible time intervals for direct radio communication with the station increased.

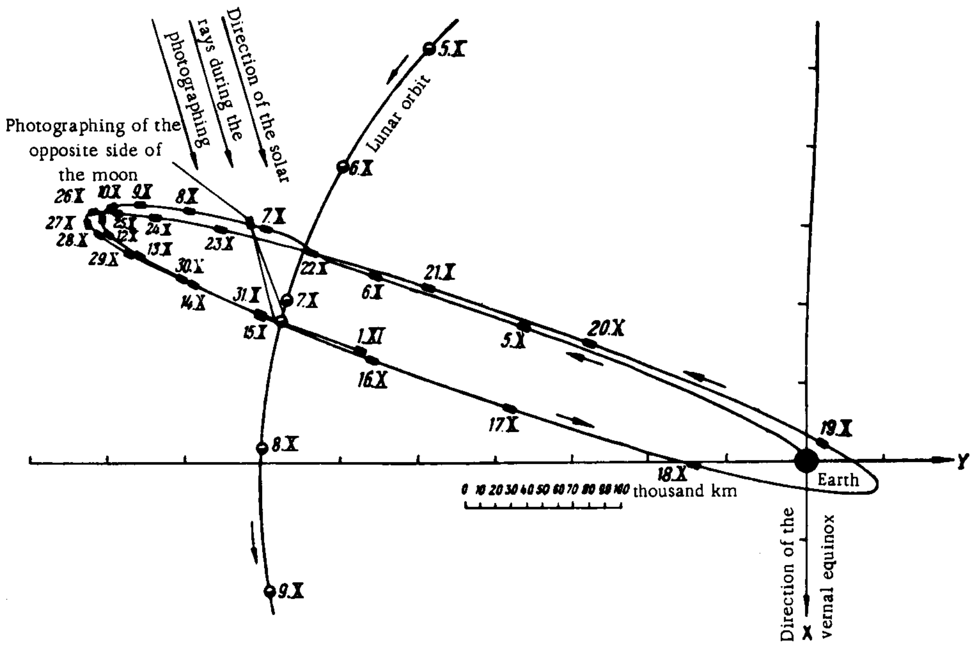

FIGURE 63. Projection of the orbit of the AIS on the plane of the terrestrial equator. The positions of the AIS and of the moon on the orbits are given for 0300 hours Moscow time each day

In returning to the earth on the first revolution, the station approached to a distance of 47,500 km, i.e., outside the dense layers of the terrestrial atmosphere, and continued its flight along an elongated orbit, nearly elliptical. The greatest distance of the station from the earth was 480,000 km.

Due to the perturbing action of the sun, the height of the orbit's perigee gradually decreased and after several revolutions, in returning to the earth, the station had to enter the dense layers of the atmosphere and was destroyed.

Artificial satellites of the moon, like those of the earth, make it possible to obtain valuable scientific information on our nearest cosmic neighbor. In the following example we show the possibility of creating an artificial satellite of the moon, in principle.

Suppose a spaceship moves along an elliptical trajectory (Figure 64), so that, at apogee on the boundary of the moon's sphere of action, its geocentric velocity is approximately 0.2 km/sec. At this point the ship's velocity increased by means of a rocket engine to 0.8 km/sec. As can be seen from the velocity triangle, the entrance selenocentric velocity becomes approximately 0.24 km/sec. This velocity and its direction make the ship move along an elliptical moon-satellite orbit. However, the orbit of such a lunar satellite leaves its sphere of action, and therefore it can be assumed that on the first revolutions the ship will no longer be a satellite of the moon.

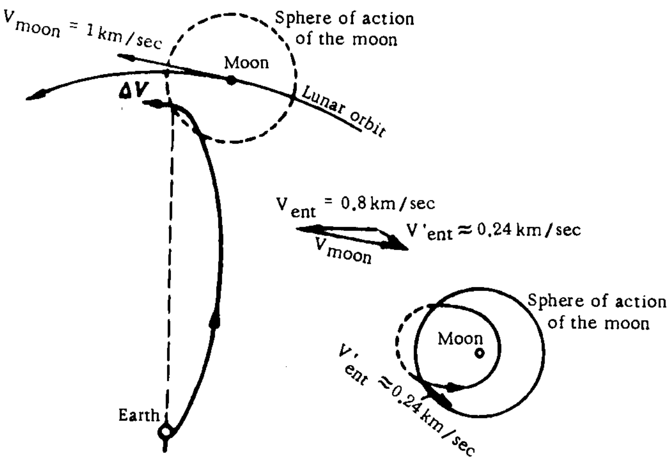

FIGURE 64. Possible scheme of launching a spaceship into an orbit of a temporary satellite of the moon; ΔV is the additional velocity given by the engine's thrust

Other ways of solving the problem of the creation of an artificial satellite of the moon are possible. For example, it is possible to create a satellite moving along a circular orbit on the boundary of the lunar sphere of action. The necessary launching conditions and required velocity movement on the boundary of the lunar sphere of action are readily determined.

We now consider briefly the conditions and trajectories for take-off from the moon in order to return to the earth. In order to reach the boundary of the sphere of action of the moon, a spaceship starting from its surface should be given a velocity of 2.344 km/sec. But with this initial velocity the ship will not return to the earth, because it will reach the boundary of the lunar sphere of action with zero selenocentric velocity. But since in this case the velocity of the spaceship with respect to the earth (its geocentric velocity) will be equal to the orbital velocity of the moon, 1.02 km/sec, the ship will become an earth satellite.

Even take-off from the moon with the required escape velocity does not make it possible to return to the earth. In this case, the selenocentric velocity on the boundary of the lunar sphere of action will be 0.385 km/sec, which is considerably lower than the oribtal velocity of the moon.

The simplest solution of the problem of return to the earth will be when the ship reaches the boundary of the lunar sphere of action with a selenocentric velocity equal in magnitude to the orbital velocity of the moon, but pointing in the opposite direction. The take-off velocity from the surface of the moon required for this purpose is 2.556 km/sec, and the geocentric velocity on the boundary of the lunar sphere of action becomes zero. The ship will move along a radial trajectory towards the center of the earth (Figure 65). The "fall" of the spaceship will last for five days, and it will reach the surface of the earth at a velocity of 11.1 km/sec.

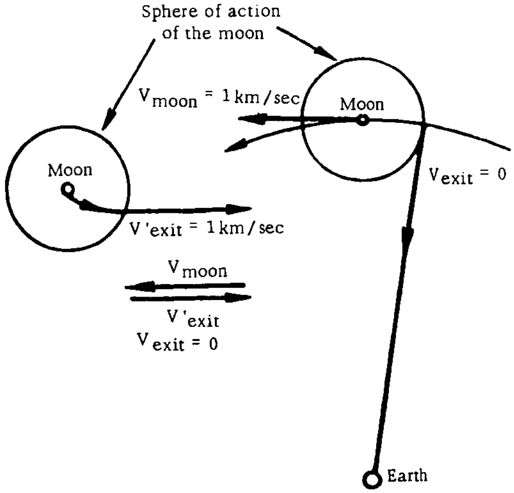

FIGURE 65. Radial return trajectory to the earth

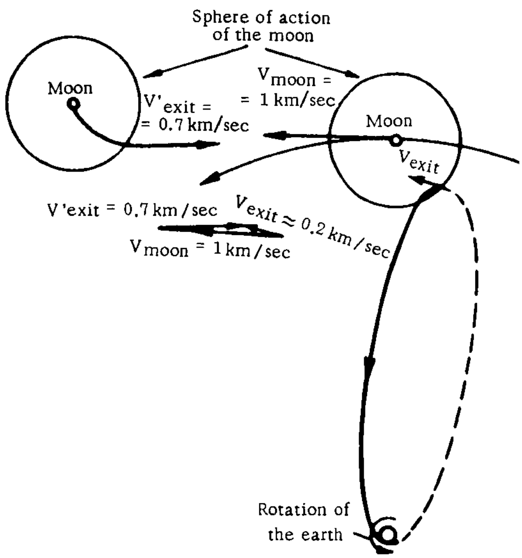

FIGURE 66. Elliptical return trajectory to the earth (one type)

More convenient return trajectories requiring, by the way, smaller energy expenditures, are also possible. One of them is an elliptical trajectory (Figure 66) with a take-off velocity from the surface of the moon equal to 2.5 km/sec. In this case, the exit selenocentric velocity will be 0.7 km/sec, and the geocentric velocity, 0.2 km/sec. Besides the lower energy expenditures, this trajectory is more advantageous from the viewpoint of soft landing of the spaceship.

Other trajectories are also possible. Such geocentric trajectories can be elliptical, parabolic, or hyperbolic orbits intersecting the surface of the earth. But the last two require large energy expenditures, and, more important, do not provide soft landing on the earth.

§8. Motion of Spaceships with Respect to the Earth and Stars

Let us now consider the peculiarities of the motion of spaceships with respect to the surface of the earth and to the stationary stars. These problems are important for the navigation of spaceships.

In order to navigate ships, determine their equatorial coordinates, find the landing region, and solve other important problems, it is necessary to know the laws of motion of spaceships with respect to the earth and the stationary stars.

Considering the motion of spaceships with respect to the surface of the earth, it is necessary to bear in mind that the unperturbed orbit of a spaceship is a plane curve, stationary with respect to the stars, and that the earth rotates about its axis with an angular velocity of approximately 15 deg/hr.

The trajectory of a satellite of the earth with respect to the earth's surface can be expressed by the relationship:

$$\tan\varphi = \sin\left(\omega'_E T + \Delta\lambda\right)\tan i,$$

where φ is the instantaneous latitude of the satellite; ω'_E is the sum of the angular rotation velocity of the earth and the mean precession velocity of the orbit; T is the time of flight of the satellite from the ascending node of the orbit to the given point of the terrestrial surface; $\Delta\lambda$ is the difference in longitudes between the position of the satellite and the ascending node of the orbit.

It is obvious that the maximum latitude, equal to the inclination of the orbit ($\varphi = i$), will occur when $\sin(\omega'_E T + \Delta\lambda) = 1$. Thus, the trajectory of a satellite lies on the surface of a spherical belt of the earth, bounded by the parallels with the latitude $\varphi = \pm i$.

"Vostok" type satellites and spaceships, whose orbits had an inclination of 65°, moved within the latitudes ± 65°, and the satellites launched in 1962 and 1963 with an orbital inclination of 49° moved within the latitudes ± 49°.

The orbit of a satellite is stationary with respect to the stars, and the diurnal rotation of the earth is from west to east. Consequently, each successive loop of the trajectory of the satellite on the surface of the earth is displaced to the west (Figures 67 and 68). It is easily seen that this displacement with respect to longitude, $\Delta\lambda$, can be found by multiplying the sum of the angular rotational velocity of the earth and the precessional velocity of the orbit by the revolution period of the ship:

$$\Delta\lambda = \omega'_E P.$$

One may also arrive at this conclusion by analyzing the above-given formula which determines the trajectory of a satellite with respect to the surface of the earth.

The "trajectory" of stars is determined only by the rotation of the earth, as a result of which each star moves along a parallel whose latitude is equal to the declination of the star (Figure 68).

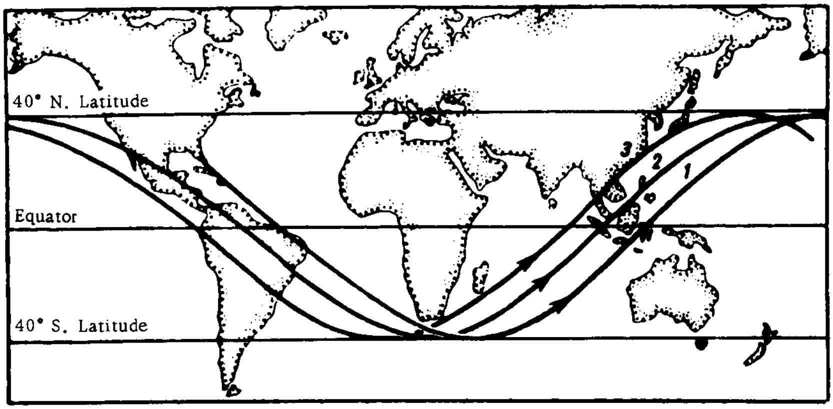

FIGURE 67. Trajectory of a satellite with orbit inclination of 40° with respect to the surface of the earth

1, 2, and 3- the first, second, and third loops of the trajectory, respectively.

[N.B. — This is clearly a plot of an American launching from Cape Kennedy, although this is not acknowledged in the Russian text.]

As an interplanetary ship gets further from the earth, its angular velocity with respect to the earth decreases, and therefore at very great distances from the earth the trajectory of a spaceship with respect to the surface of the earth will be determined, like the trajectories of stars, only by the earth's diurnal rotation. On the terrestrial surface it will coincide with the parallel whose latitude is equal to the declination of the spaceship at the given time (Figure 69). This coincidence, as can be seen from the figure, is greater the farther the ship from the earth.

The position of the spaceship at a given moment with respect to the stars is determined by its equatorial coordinates, the right ascension α and the declination δ. The apparent motion of a ship among the stars can be very complicated, since it is determined both by the motion of the ship and by the motion of the earth in its orbit with respect to the sun (Figure 70). As seen from the figure, with respect to the sun, the spaceship has a velocity higher than that of the earth, and therefore in equal time intervals it traverses larger distances along its orbit. An observer on the earth projects the ship on to the celestial sphere. First he sees the ship moving in one direction, then in the opposite direction (retrograde motion). Such motion with respect to the stars is, by the way, characteristic also of the planets (Figure 71). The retrograde motion of Mars, for example, lasts for 70 days. The arc of its retrograde motion on the celestial sphere is about 16°. Mars makes such loops in its apparent path among the stars once in 780 days, Jupiter, once in 399 days, and Saturn, once in 378 days.

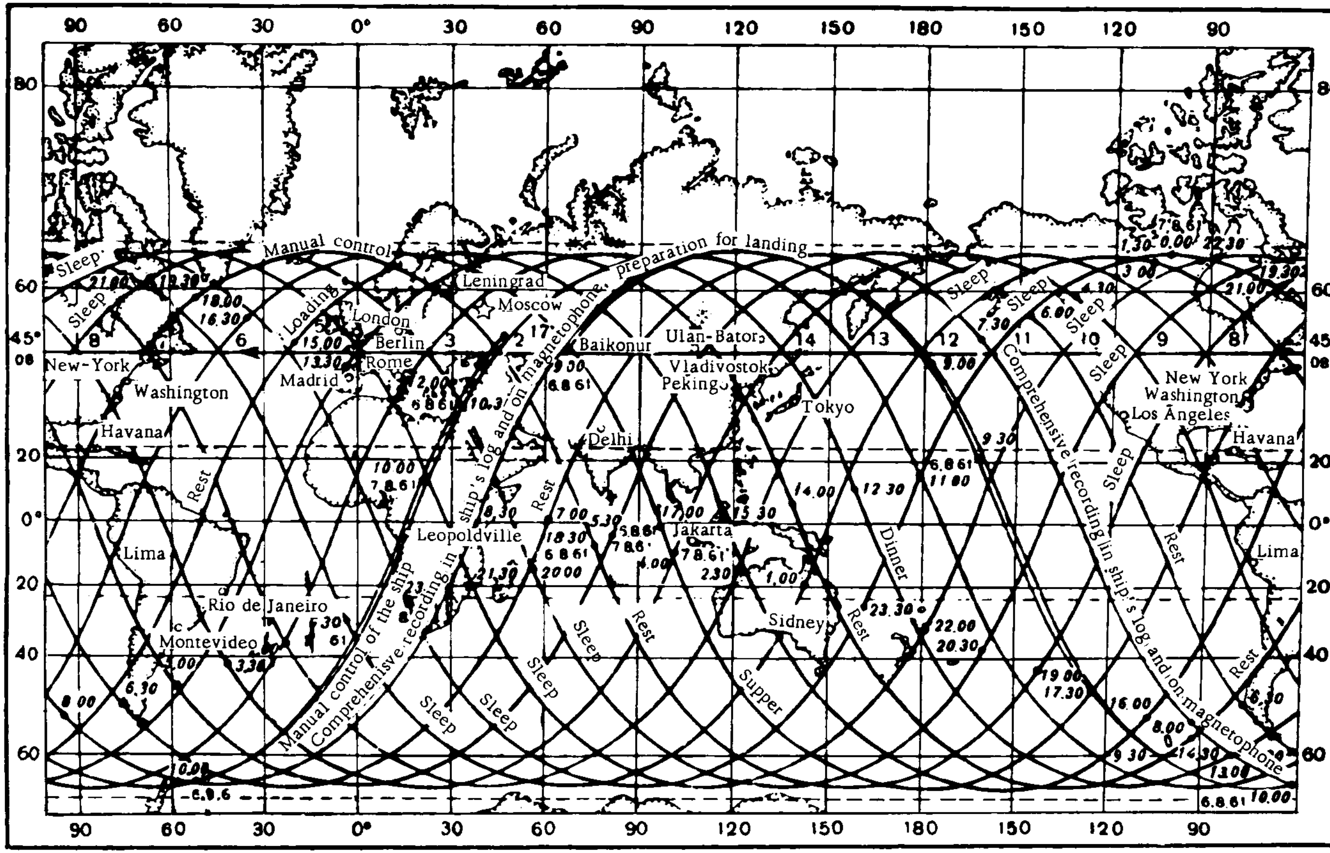

FIGURE 68. Trajectories with respect to the surface of the earth of the satellite "Vostok II" and of the star Deneb (α Cygnus), whose declination for the 1962 epoch is + 45°08'

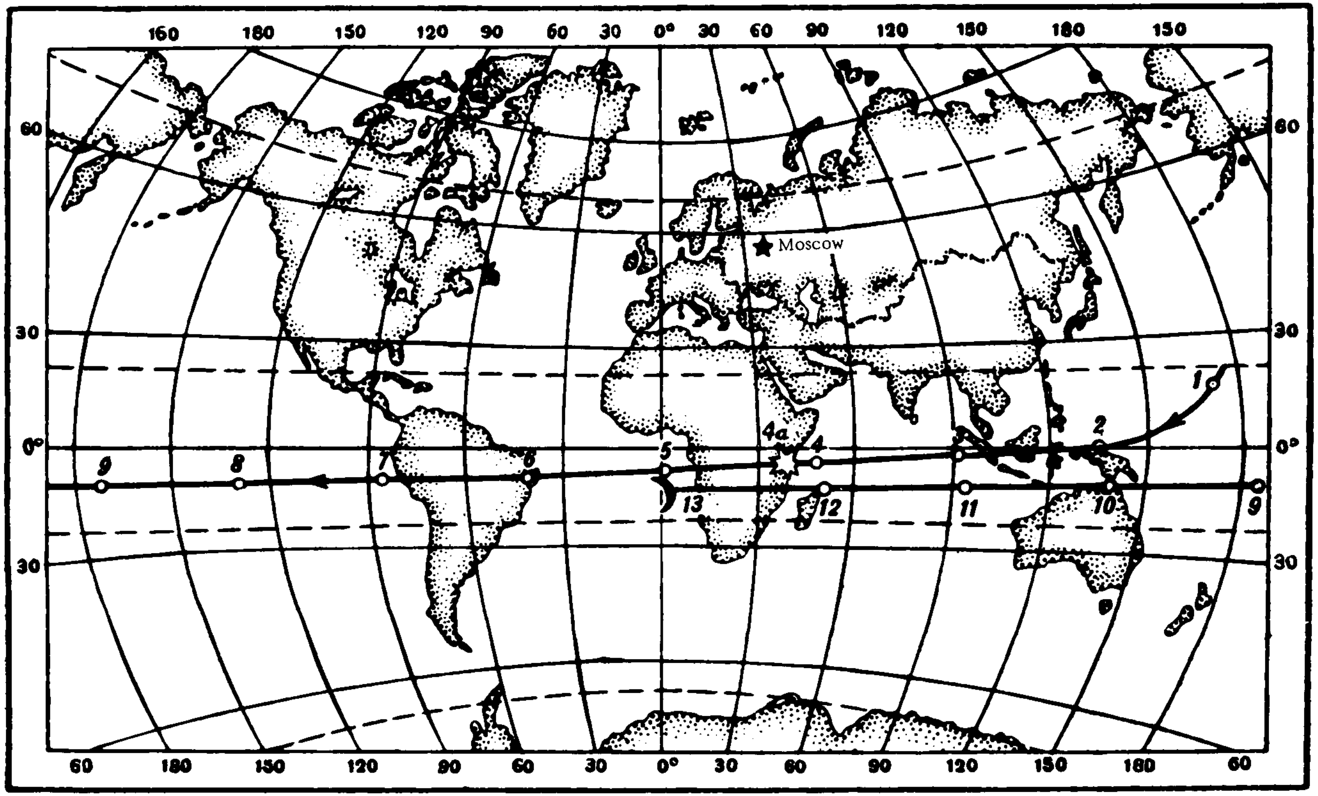

FIGURE 69. Trajectory with respect to the surface of the earth of the second Soviet cosmic rocket launched to the moon. The numbers on the figure correspond to the successive positions of the rocket's projection on the surface of the earth

1- at 12 hr 00 min, on 12 September, 4a- formation of an artificial comet; 13- at 00 hr 02 min 24 sec on 14 September – the point of encounter of the rocket with the moon at a distance of 371,000 km from the earth.

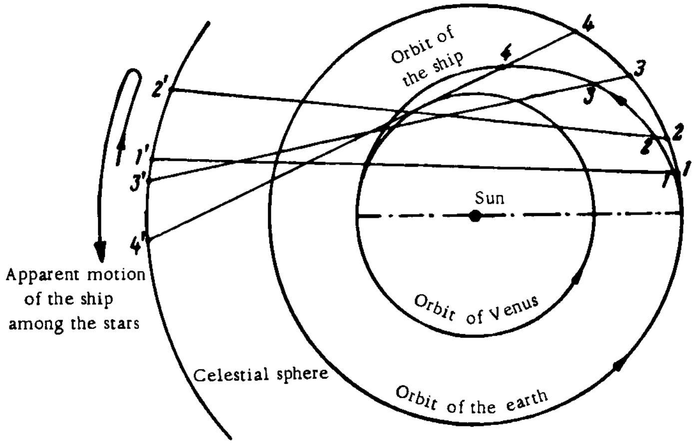

FIGURE 70. Apparent motion among the stars of a spaceship traveling to Venus

1, 2, 3, 4- positions of the earth and the spaceship after equal time intervals; 1', 2', 3', 4'- apparent positions of the spaceship among the stars (on the celestial sphere).

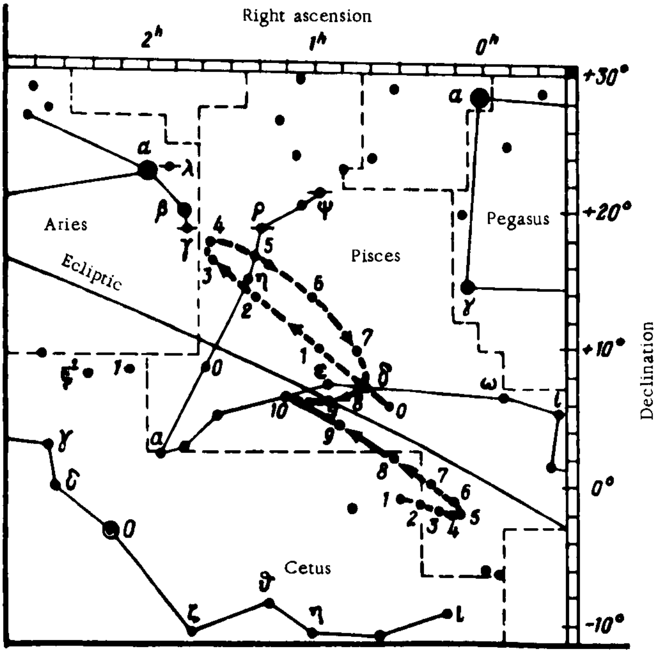

FIGURE 71. Apparent motion of Venus (the broken curve) and of the first Soviet automatic interplanetary station to Venus among the stars (the figures give the positions of the station and of Venus in 10-day intervals)

These in general are the peculiarities in the motion of spaceships with respect to the surface of the earth and with respect to the stars.

§9. To the Stars

Remote suns twinkle in the depths of the night sky. How many riddles they hide! What is the mechanism of the formation of stars and star systems? What is the source of the huge amount of energy released by the stars? What is their composition? What are the reasons for the periodic variation of the color of certain stars? What is the reason for the fantastically high density of one of the star groups? Do stars, like the sun, have planetary systems? These are only some of the important questions asked about the stars. Whatever guesses, hypotheses, and assumptions are made can for the time being be based only on study of the radiation of stars and star systems by telescopes and radio-telescopes, and on analysis of the variation of their positions. The difficulties of studying stars are due to their huge distances.

Automatic stellar stations and spaceships will be the only means of bringing distant stars nearer to a terrestrial observer. They are the only means of obtaining the large amount of objective scientific information required to reveal the secrets of the stars.

Even now, after the flight of the first astronauts, flight to the stars seems a fantastic dream. But only a few years ago, flights of man around the earth on satellites, flights to the moon, and flights of automatic interplanetary stations to Venus and Mars seemed equally fantastic.

Mankind will hardly restrict itself to study of the solar system. The time will arrive when powerful rocket systems will launch spaceships into orbits to the near stars. The recent astonishing successes of science and engineering instill confidence that this time will arrive and that the very complicated problem of flight to the stars will be solved by man's genius.

It is still impossible even to list all the difficulties which scientists and engineers will encounter in solving the problem of a stellar flight, but it can be confidently asserted that most of them are due to their remoteness. The first question to be asked is whether man's lifetime is sufficient to fly to one of the nearest stars and return to the earth? What should be the minimum velocity of a starship which is to visit the nearest stars? Let us consider this problem in more detail.

Table 17 gives a list of the closest bright stars, situated at a distance of no more than 10 ps (the distances to these stars are given in parsecs and light years).

Even a superficial look at the table leads to the conclusion that, if flight to the stars is to be possible, the velocities of the spaceships intended for this purpose must be tremendous, close to the velocity of light. But, as is known, at velocities near the velocity of light, time is slowed down in the moving reference system. The velocity of light is a constant quality, and therefore with moving clocks the time units become longer, and the length units become shorter in the direction of the motion. This should be taken into account in the calculations of future astronauts.

TABLE 17. Stars nearest to the earth and times of flight to them and back (flight duration by shipboard clocks is 25 years)

Star designation	Distance to star		Required ratio of velocity of starship to velocity of light	Time of flight to star and back by terrestrial clocks, in years	Difference in flight times between terrestrial and shipboard clocks	Minimum time for obtaining information, in years
	in parsecs	in light years				
α Centauri	1.32	4.31	0.3254	26.5	1.5	17.56
α Canis Majoris (Sirius)	2.68	8.74	0.5734	30.5	5.5	23.99
ε Eridani	3.30	10.76	0.6521	33.0	8.0	27.26
τ Ceti	3.32	10.83	0.6563	33.0	8.0	27.33
α Canis Minoris (Procyon)	3.44	11.20	0.6673	33.6	8.6	28.00
α Aquilae (Altair)	4.88	15.90	0.7861	40.5	15.5	36.15
η Cassiopeiae	5.49	17.91	0.8230	43.5	18.5	39.66
δ Pavonis	5.95	19.40	0.8406	46.2	21.2	42.50
β Hydri	6.58	21.45	0.8640	49.7	24.7	46.35
α Piscis Austrini	6.90	22.48	0.8742	51.4	26.4	48.18
ξ Ursae Majoris	7.69	25.08	0.8954	56.0	31.0	53.08
π Orionis	7.81	25.47	0.8977	56.7	31.7	53.82
χ Draconis	8.20	26.72	0.9056	59.0	34.0	56.22
γ Leporis	8.20	26.72	0.9056	59.0	34.0	56.22
α Lyrae (Vega)	8.26	26.94	0.9072	59.4	34.4	56.64
μ Herculei	8.55	27.86	0.9126	61.1	36.1	58.41
η Bootae	8.93	29.11	0.9188	63.4	38.4	60.81
δ Eridani	9.17	29.91	0.9226	64.9	39.9	62.36
β Virginis	9.90	32.28	0.9326	69.2	44.2	66.88
β Geminum (Pollux)	10	32.60	0.9337	69.8	44.8	67.50
ζ Herculei	10	32.60	0.9337	69.8	44.8	67.50

[N.B. There are some minor inaccuracies in Table 17. As will be seen, the last column should equal the third plus half the fifth.]

Thus, the flight duration to a star according to the shipboard clocks and to terrestrial clocks will be different. The following relationship exists between the flight time t according to terrestrial clocks and the flight time t_0 according to shipboard clocks:

$$t = \frac{t_0}{\sqrt{1-k^2}},$$

where k is the ratio of the flight velocity of the starships to the velocity of light.

If the distance to the star is S light years, then the time of flight to the star and back according to terrestrial clocks will be:

$$t = \frac{2S}{k}.$$

Substituting this value for the flight time t in the previous formula, and solving it for k, we obtain

$$k=\frac{2S}{\sqrt{t_0^2+4S^2}}.$$

By this formula it is possible to calculate the value of the coefficient k, for which a starship will fly to a given star and back in a time t_0 according to shipboard clocks. From k, as obtained by this formula, the corresponding velocity $V=kc$ can be computed.

In the first approximation, without dwelling on a detailed analysis of this problem, we will consider that man has at his disposal 25 years of his life for the flight. Besides the time for the flight, time is also required for training, for data processing after the flight, and for rest. Table 17 shows that even flight to the nearest star, α-Centauri and back, which would take 25 years by shipboard clocks, and 26.5 years by terrestrial clocks, requires a velocity of 0.3254c. Crew members of such a starship, upon returning to the earth, will be one and a half years younger than men born on the same date who never left the earth.

For more distant stars, as can be seen from the table, the required velocity, the flight duration by the terrestrial clocks, and the difference between them and shipboard clocks all increase.

Even by shipboard clocks a duration of 25 years for such an "experiment" is very long. Flight to stars situated at a distance of 10 ps and subsequent return to the earth will occupy 69.8 years for people remaining on the earth. Is it possible to reduce the flight durations and periods that must pass before scientific information can be obtained from the flight? This can be done, in the first instance, by increasing the velocity of the starship. In principle this measure is extremely effective for shortening the duration of flight to the stars. However, it is hardly necessary to demonstrate the complexity of the problem of obtaining velocities near that of light. It is made more difficult by the fact that with increasing velocity the mass of the spaceship increases, and this in turn increases the energy expenditures necessary for further increments in velocity. This is particularly true at velocities near that of light. In the case of a 1½-ton spaceship flying at a velocity of 11.2 km/sec, the mass increases by only one milligram, but the mass of a proton whose velocity is increased by a factor of 27 thousand in an accelerator increases by a factor of more than 10 over its rest mass /9/.

The duration of a star experiment can also be shortened, if observations are transmitted to the earth by the crew of the starship by radio immediately after reaching the star. In this case the scientific information will be received after a time equal to the flight time to the star by the terrestrial clocks plus the distance from the star to the earth in light years (see Table 17). It should be borne in mind, however, first, that this method requires the solution of the problem of communications over tremendous distances, and second, that it is hardly possible to transmit from the ship to the earth all the scientific information obtained by radio. In addition, as can be seen from the table, the effectiveness of this measure is smaller, the greater the distance to the star.

The theory of relativity therefore gives a positive answer to the question of whether man can reach stars which are situated at distances considerably larger in light years than the time he has for the flight. But even without touching on the tremendous technical difficulties connected with the attainment of velocities close to that of light, many other aspects of this problem are unclear. First of all, of course, is the problem of man's ability to live at velocities near the velocity of light. A complete answer to this problem can be given only by corresponding experiments with living organisms. Scientists are already thinking about this.

One of the main purposes of flights to stars will no doubt be the search for living organisms, particularly in their higher forms, in the depths of the universe. In this connection, an estimate of the probability of detection of higher forms of life by starships is of interest.

In the first chapter we said that according to tentative calculations of some astronomers, for each million stars there should be on the average one inhabited planetary system. Consequently, the probability of observing inhabitants in the depths of the cosmos in a single flight to the stars is according to these figures very low, $P_1 = 10^{-6}$.

On the basis of this probability, even if the number n of flights is quite large, the probability of obtaining a positive result is negligible (Table 18).

TABLE 18. Probability of one or more encounters with an inhabited planet in flights (probability of encountering a single flight $P_1 = 10^{-6}$)

n	2	3	4	5	10	20	50	100
P_n in %	0.0004	0.0006	0.0008	0.001	0.002	0.004	0.01	0.018

Let us now calculate the number of flights which is required to guarantee a probability P_n of one or more encounters with an inhabited planetary system. This probability can be solved by the formula

$$n = \frac{\lg (1 - P_g)}{\lg (1 - P_1)},$$

some values of P_g calculated by it are given in Table 19.

TABLE 19. Number of flights of a starship guaranteeing a probability P_g of one or more encounters with an inhabited planetary system

P_g	n	P_g	n	P_g	n
0.1	53,200	0.5	350,000	0.90	1,162,800
0.2	112,700	0.6	462,000	0.95	1,152,600
0.3	180,100	0.7	608,000	0.98	1,975,600
0.4	258,000	0.8	812,800	1.00	∞

Thus, if we assume that there should be one inhabited planetary system per one million stars, the probability of encountering one such system even in a hundred flights amounts to only 0.018%, and in order to have a 90% probability of encountering an inhabited planetary system a tremendous number of flights — 1,162,800 ! — is necessary. Many years would be necessary for so many flights, even if starships were launched daily.

These pessimistic conclusions, however, are based on the highly tentative assumption of scientists that there is only one inhabited planetary system per one million stars. It is perfectly possible that inhabited planetary systems are more often encountered in the universe. Latest data indicate that inferior forms of life may be encountered in the universe quite often. In addition, scientists are already at work on the problem of detecting inhabited planetary systems from the earth. Maybe we will even succeed in observing, from the earth, planets inhabited by intelligent beings. It will then be unnecessary to make such a huge number of flights for man to meet beings similar to him, who live near other suns.

Chapter III

BASIC PROBLEMS OF SPACE NAVIGATION AND METHODS OF SOLUTION

§1. Basic Problems of Space Navigation

We may in general divide the basic problems of space navigation into the following groups: selection and calculation of the ship's trajectory; analysis of the motion along the preassigned orbit; orbit correction; transfer to an orbit with respect to a celestial body; the flight target; landing.

Most of these problems have been discussed in sufficient detail in the previous chapters, and therefore we will here consider only two problems — analysis of the motion along the preassigned orbit and correction of the ship's trajectory.

These two problems are extremely important to the overall navigation of spaceships. Due to the long flight duration, even to near celestial bodies, and to the practical impossibility of accounting for all perturbations, the deviations of the actual orbit from the calculated one may be considerable, even for small errors in the calculated orbital parameters. For example, take the case of flight to Venus. The flight duration, as was shown in the previous chapter, is from 2 to 5 months. An accumulated time error of only 10 minutes at the calculated exit point may cause a deviation in the actual entry point into Venus' sphere of action of more than 21,000 km. This deviation amounts to more than 3% of the radius of Venus' sphere of action. Such an uncorrected deviation may considerably modify the motion of the spaceship within Venus' sphere of action as compared with the calculated motion. Even this very simple example shows the need of solving the problem of analysis and correction of the trajectory of a spaceship.

In addition, it should be borne in mind that any phenomena or scientific data observed by the crew of a spaceship should be "fixed" in space. In other words, the crew should note not only the time of the phenomenon or of the scientific observation, but also the observed location and the ship's coordinates at that moment. Consequently, it becomes necessary periodically to determine the actual coordinates of the ship.

Finally, the analysis of the ship's motion, and in particular the determination of its position, makes it possible to calculate the elements of the actual orbit along which it moves. This is also important, since it makes possible not only controlling entrance into the precalculated orbit, but also calculation of the ephemeris of the ship, i.e., prediction of its further motion.

Which elements should be determined in the analysis? First, the spatial coordinates of the ship must be determined, but this is insufficient. In general, the purpose of space flight is to reach other celestial bodies moving at high velocities in space. It is possible to carry out the flight rigorously along the preassigned trajectory and not to fall into the sphere of action of the destination planet. The latter may occur when the flight regime is not followed with respect to time. Consequently, timing of the motion along the preassigned trajectory must also be analyzed.

It is true that the error in the arrival of a spaceship at a calculated point will be mainly determined by the timing errors of the launch or the transfer from one orbit to another. All the remaining causes will probably not result in large errors. However, as was already shown in the example above, even small errors in the time of arrival at calculated points of the trajectory may considerably modify the subsequent motion of the spaceship and make it more difficult to reach the target.

Naturally such an important motion parameter as the velocity of the spaceship or its components must also be controlled. This is so obvious that it does not require any explanations. In certain cases a need may also arise for determining in flight the accelerations of the ship and their components.

One of the basic methods for locating the path of an airplane or ship is based on the determination of its position lines. The position line of an airplane is a line on the surface of the earth of possible projections of the airplane. For an airplane situated above any of its points, the measured parameter is constant. That is, the position line is the locus of points corresponding to the measured value of the parameter. The airplane is always situated over one of its points at the moment of measurement of the given parameter.

The geometrical form of the position line is determined by the parameter being measured. For example, the distance from an airplane to a ground station is measured by a radio system to be 185 km. We cannot determine from this the coordinates of the airplane at the given moment, but we do know the airplane is over one point of a circle on the surface of the earth with a radius of 185 km and center at the ground station. In this case, the position line is a line of equal distances, a circle.

To the measured course angle at the ground station corresponds a line of equal azimuths, a line on the surface of the earth at all points of which the azimuth to the station is constant. The azimuth of an airplane or ship measured from the ground by means of a radio direction finder corresponds to a great circle on the surface of the earth.

Thus, in order to determine the position line of an airplane or ship it is necessary to measure some parameters, and the location of the airplane or ship is the intersection point of two position lines, corresponding to the two parameters measured simultaneously. Comparison of the position obtained this way with the precalculated one determines the deviation of the airplane or ship from course. Comparison of the time at which the position was determined with the precalculated flight schedule makes it possible to find out how the flight plan is maintained. Finally, from the distance between two positions of the airplane and the travel time, the actual velocity of the airplane can be easily determined.

This method can also be used in space navigation, but in this case the measured parameter corresponds not to a position line, but to a position surface. For example, suppose a distance from the spaceship to the earth is one million km. The crew of the ship knows that the ship is situated at one of the points of space lying one million km from the earth. All these points form one position surface of the spaceship, a sphere whose center is at the center of the earth, and whose radius is one million km.

What parameters may be used to determine the position surface of spaceships? They may include the distance to a celestial body, the earth, the moon, the sun, and so on, measured by radio-electronic methods from the spaceship, or the distance from the earth to the ship, measured from the earth. It is also possible to measure optically from the ship a parameter of a celestial body such as its apparent angular diameter.

In order to determine the position surface one can also measure the angle between the lines from the ship to the centers of two planets, the centers of a planet and the sun, or a star and the center of a planet or the sun. The parameter determining the position surface of the spaceship can also be some direction from the surface of the earth to the ship, measured by radio-electronic or optical methods. These might include the azimuth of the spaceship and its height above the horizon. It is possible to measure the Doppler shift of the frequency of the signals received on the earth from the spaceship's radio, as well as the shift of the signals emitted by a radio transmitter on the earth. A position surface of a spaceship can also be obtained by measurement of the phase difference between signals of the ship's radio transmitter as received by two radio stations situated some distance apart on the earth.

It is also possible to measure some other parameters, or to use sums or differences of some of the above-indicated parameters in order to determine another position surface of the ship. These might include sums or differences of the distances to two celestial bodies and sums or differences of the angular diameters of two celestial bodies.

§2. Position Surfaces of Spaceships

Let us now determine the geometrical form of the position surfaces corresponding to the above parameters.

As is already known, the distance to a celestial body measured from a spaceship, or the distance to the spaceship measured from the earth corresponds to a spherical position surface (Figure 72a), whose center is at the center of the celestial body (the earth). If the measured distance is D, and the coordinates of the center of the planet (the earth) in the selected rectangular coordinate system are X, Y, Z, then the equation of the sphere will have the form:

$$(x-X)^2+(y-Y)^2+(z-Z)^2=(R+D)^2,$$

where R is the radius of the planet.

For spaceships which fly near the earth, it is expedient to choose the origin of the coordinate system at the center of the earth. In this case when measuring the distance to the earth, the equation of the sphere is

$$x^2+y^2+z^2=(R+D)^2.$$

It can be easily shown that in the case of measuring the apparent angular diameter of a planet or the sun, the position surface will also be a sphere (Figure 72b), since the apparent angular diameter of the celestial body is the same at all points equidistant from its surface. The equation of the position in this case is

$$(x-X)^2+(y-Y)^2+(z-Z)^2=\frac{R^2}{\sin^2\frac{\beta}{2}},$$

where β is the angular diameter of the celestial body (the sun).

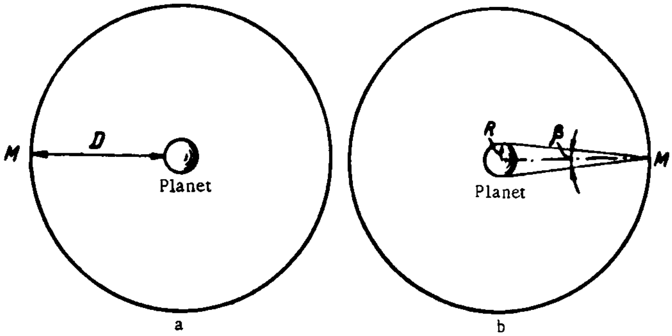

FIGURE 72. Spherical position surface of a spaceship determined
a- by the distance D to the planet; b- by the apparent angular diameter β of the planet.

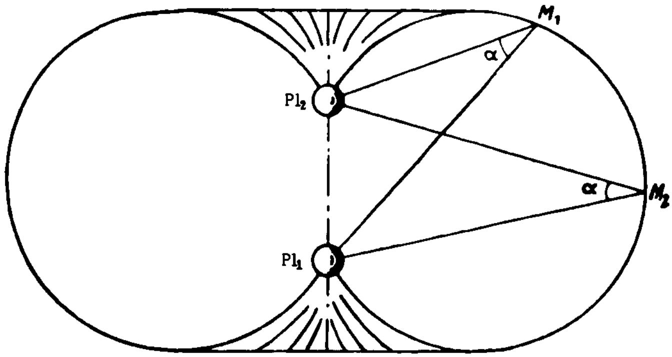

FIGURE 73. Cross section of position surface (cyclide) of a spaceship
α- the angle between the centers of the two planets; M_1 and M_2 - different positions of the ship.

Let us now assume that the angle α between the centers of two planets Pl_1 and Pl_2 is measured from the spaceship (Figure 73). In the plane of the figure, the position line corresponding to the measured parameter α will be circular arcs passing through the centers of the planets, since the planets are subtended at the same angle from each of their points. In space, the position surface is obtained by rotating the arcs of these circles with respect to the axis Pl_1-Pl_2. This type of surface is called cyclide. Its equation in a bipolar coordinate system can be represented in the form:

$$l^2=R_1^2+R_2^2-2R_1R_2\cos\alpha,$$

where l is the distance between the centers of the planets calculated by the formula

$$l^2=(X_2-X_1)^2+(Y_2-Y_1)^2+(Z_2-Z_1)^2,$$

R_1 and R_2 being the distances from the ship to the respective planets determined by the relation

$$R_i^2=(x-X_i)^2+(y-Y_i)^2+(z-Z_i)^2.$$

Let us now consider the form of the position surface formed by the angle between the line of centers of planet and spaceship and that of planet and star. We shall use a rectangular coordinate system whose origin is at the center of the planet. The Z axis will coincide with the planet-star line (Figure 74). Assuming that the spaceship is at point M, $\alpha = \measuredangle MOZ$, is the angle between the two lines of centers. Because the stars are so distant, light rays from them are received in parallel beams, and therefore the angle will be constant at all points of a circular conical surface. The vertex of the surface is situated at the center of the planet, and its axis coincides with the striaght line from the planet to the star. In the selected coordinate system the equation of a conical surface can be represented in the form

$$\frac{x^2+y^2}{z^2}=\tan^2\alpha.$$

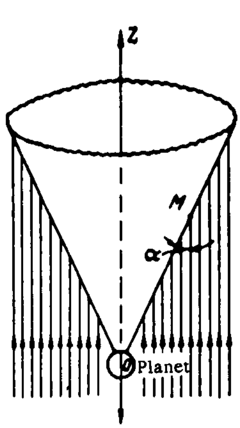

FIGURE 74. The position surface of the spaceship as a circular cone

α - the angle between the ship-planet and starship lines (spaceship at the point M).

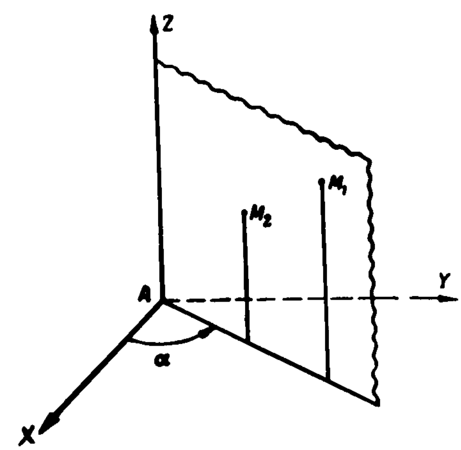

FIGURE 75. Plane position surface of a spaceship

α- the azimuth of a ship situated at the point M_1 or M_2.

Let us now determine the geometrical form of the position surface obtained when the earth is the reference point for location of the spaceship. Suppose a direction finder is situated at a point A on the earth (Figure 75).

It is obvious that to the measured azimuth α of the ship corresponds a position surface which is a plane, containing the AZ axis. The equation of a plane in the selected coordinate system is:

$$\tan \alpha = \frac{y}{x}.$$

When the elevation of the spaceship above the horizon, or its altitude angle is measured from the earth, the position surface will be a circular cone whose vertex is at the point where the elevation measurement was made, and whose axis is along the local vertical.

Next, let us consider the position surfaces which are obtained by measuring the Doppler shift of the received signal frequency. The Doppler effect is the variation of the frequency of a signal received when the transmitter or receiver is moving. Doppler frequency shift, or Doppler frequency, is the term applied to the difference between the frequency f_{rec} of the signals received and the frequency f_{tra} of the signals transmitted:

$$F_d = f_{\text{rec}} - f_{\text{tra}}.$$

Suppose the transmitter is mounted on a spaceship, and the receiver is at the point A on the earth (Figure 76). The Doppler shift of the received signal in this case is given by the relation:

$$F_d = \frac{f_{\text{tra}}}{c} \bar{V} \cos \beta,$$

where c is the propagation velocity of the radio waves, and $\bar{V}$ is the velocity of the ship.

For constant f_{tra} and $\bar{V}$, as can be seen from the formula, the frequency shift depends only on the angle β between the velocity vector and the direction to the receiver. Consequently, at all points of space with the same value of the angle β, the Doppler frequency shift will be the same. But points with equal values of the angle β form coaxial circular conical surfaces whose common vertex is at the transmitter and whose axis is along the velocity vector $\bar{V}$. Consequently, measurement of the Doppler frequency shift gives the position surface of the spaceship in the form of coaxial circular cones with a common vertex (Figure 77).

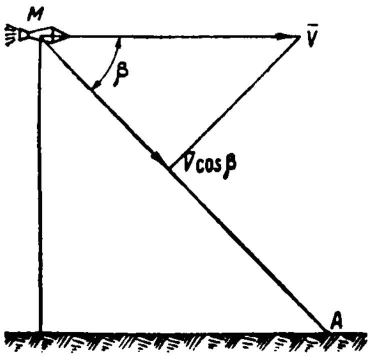

FIGURE 76. Radial component of the velocity of a spaceship measured by the Doppler shift

Notice an important feature of these position surfaces. The opening of the cone is determined by the angle β. On the straight line coinciding with the velocity vector ($\beta = 0°$) the Doppler frequency shift is maximum. The conical position surface degenerates here to a straight line. For $\beta = 90°$, $F_d = 0$. Consequently, zero Doppler shift corresponds to a plane position surface which is perpendicular to the velocity vector of the ship.

Thus, a family of conical position surfaces is obtained which move together with the ship in space. The larger the frequency shift, the smaller the opening of the cone. In extreme cases the conical surfaces degenerate into a straight line and a plane.

As can be seen from the formula, the vertex angle β of the cone is determined by two parameters: the measured value F_d of the Doppler frequency shift and the ship's velocity V. But V depends on the position of the ship in space, which is still unknown. Consequently, position surfaces which are circular conical surfaces cannot be used to determine the position of a spaceship when the velocity is unknown. For this purpose we can use only a plane, which corresponds to the unique case $F_d = 0$.

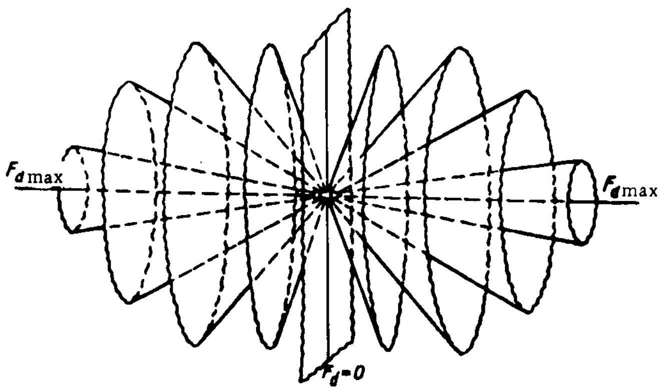

FIGURE 77. Position surfaces which are obtained when measuring the Doppler frequency shift

Another feature of these position surfaces should be remembered. All our previous considerations were based on the fact that the receiver is stationary in inertial space. However, in reality this is not so. A receiver which is situated on the earth has some linear velocity due to the diurnal rotation of the earth. As was shown in the first chapter, this velocity (in km/hr) is equal to 0.465 cos φ. This velocity is directed along the parallel to the east.

The Doppler frequency shift is due to the relative velocity of the transmitter (with respect to the receiver). It is obvious that it will not be equal to the absolute velocity of the spaceship, on which the transmitter is mounted. The relative velocity is the difference between the velocity vector of the ship and the linear velocity vector of the point on the earth where the receiver is situated.

Thus, for a given location of the receiver, the position surfaces will be symmetric with respect to an axis coinciding with the vector of the relative velocity of the spaceship. In other words, the direction of the vector of the relative velocity will be the axis of conical position surfaces, and the extreme position surface, a plane, will be perpendicular to this vector. For other locations of the receiver, the orientation of the position surface, or plane, in space will differ corresponding to different orientations of the relative velocity vector.

When measuring the Doppler frequency shift of the signal received on the spaceship, the vertices of the conical position surfaces will coincide with the location of the radio station on the earth. In this case too, if the velocity of the spaceship is unknown one may use only the plane to determine the position.

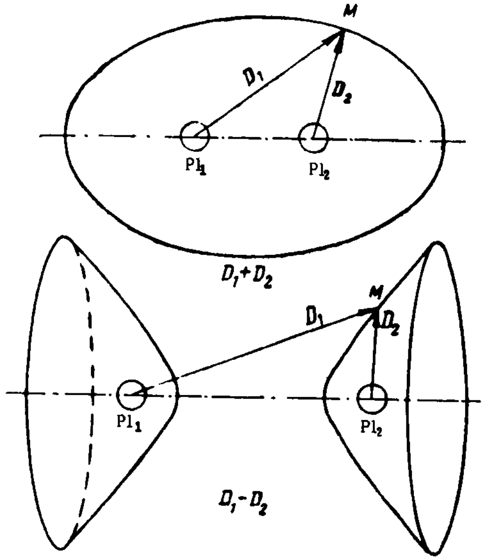

FIGURE 78. Position surfaces of a spaceship as an ellipsoid and a hyperboloid of revolution

D_1 and D_2- distances from the centers of the planets to the ship at the point M.

The velocities of spaceships are sufficiently high for the Doppler frequency shift to reach several Mcps.

Determining position surfaces by measuring the phase difference of the signals from a spaceship's transmitter as received by two ground stations leads to hyperbolic position surfaces (hyperboloids of revolution). Their axes coincide with the straight line connecting the ground stations. At the mean distance between the receiving stations the hyperboloid degenerates into a plane perpendicular to the base line.

If the measured parameter is, for example, the sum of the distances to two celestial bodies, the position surface of the spaceship will be an ellipsoid of revolution with an axis passing through the centers of the celestial bodies Pl_1 and Pl_2. The difference between the distances to two celestial bodies corresponds to a hyperboloid of two sheets whose rotation axis passes through the centers of the celestial bodies (Figure 78).

§3. Principles for Solution of the Problems of Analysis and Correction of the Spaceship's Trajectory

The solution of the problem of controlling the motion of a spaceship requires determination of its position at a given time, as well as the determination of its actual orbital elements. The actual orbit of a spaceship can be determined either by two successive positions in space, or from three directions from the earth to the ship, obtained at different times.

The directions to a spaceship, usually given in some spherical coordinate system, are obtained by the intersection of two position surfaces (two planes, two conical surfaces, a plane and a conical surface).

For interplanetary ships far from the earth, determination of the actual orbit is possible only by determining successive positions of the ship.

Let us consider the principle of the determination of the position of a ship in space.

In general, the analytic relation between the measured parameter α, the coordinates X, Y, Z of the center of the celestial body or a point of the earth at which the measurement of the parameter α is made, and the coordinates x, y, z of the spaceship can be represented:

$$\alpha = f(X, Y, Z, x, y, z).$$

This equation contains three unknown coordinates; consequently, to determine them it is necessary to measure any three parameters, or one parameter from three different points on the earth. The resulting three simultaneous equations will contain three unknowns, the coordinates of the spaceship at the given moment. The geometrical interpretation of the solution consists in finding the intersection points of three position surfaces of the spaceship.

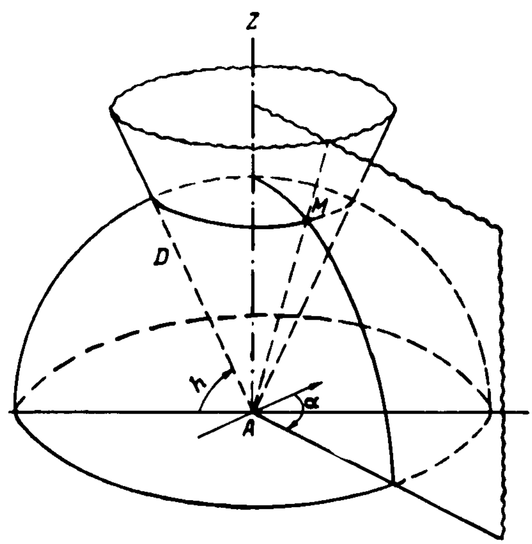

FIGURE 79. Determination of the position of a spaceship at the point M from its height h (elevation angle), azimuth α, and distance D

Let us assume that at some point A on the earth (Figure 79) there is a station which measures three parameters, the height above the horizon, or the elevation angle, the azimuth of the spaceship, and its distance.

Knowledge of these parameters makes it possible to determine the unique position of the spaceship at the given moment. The height, or elevation angle h, gives a conical surface whose vertex is at the ground station, and whose axis is along the local vertical. The distance D results in a sphere whose center is at the ground station. The intersection of these two position surfaces gives a position line of the ship. In this case it is a circle, lying at some height above the earth in a plane perpendicular to the local vertical. The intersection of this line with the third position surface, a plane, determined by the measured azimuth α, gives the point M at which the spaceship is situated at the given moment.

In this example, all three position surfaces intersect at one point, and the position of the spaceship is determined uniquely. In general, the position surfaces intersect at two, and in some cases, at four points.

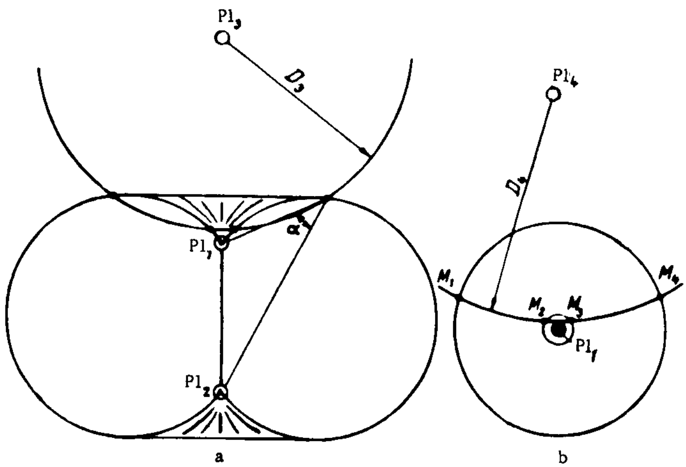

FIGURE 80. Four possible intersection points (M_1, M_2, M_3, and M_4) of three position surfaces (one cyclide and two spheres, obtained from the distances D_3 and D_4 to the planets)

a- "side" view; b- view from "above".

Suppose that in order to determine the location of a spaceship three parameters are measured, the angle between the centers of two planets and the distances to two other planets (Figure 80). The intersection of the cyclide with one of the spherical surfaces gives two position lines, and two circles are obtained. Their intersection with the second sphere gives four points: M_1, M_2, M_3, and M_4. It is obvious that only one of these points is the actual position of the ship in space at the given moment, and the remaining three points are false. Which one is real can be solved in two ways. First of all, it is most likely that the real one is the point which is nearest to the calculated orbit. Sometimes this method cannot answer the question. In fact, the points M_2 and M_3 may lie close to one another, and the calculated orbit may pass between them. In this case it is worthwhile to measure a fourth parameter, resulting in another position surface which must intersect the first three at the actual location of the spaceship.

The previous considerations were based on the assumption that the parameter determining the position surface is measured with absolute accuracy. However, in reality, any parameter is measured with some error. Let us now consider the possible measurement errors of some of the principal parameters and their influence on the solution of problems of cosmic navigation.

Several methods of determining position surfaces depend on finding the direction to a star. Equipment for doing this automatically is already available. There are for example, airplane navigational instruments, astrotrackers, which work by finding star positions. The reasons for errors in direction finding of stars are the aberration of light,* proper motion of the stars, and instrument errors.

Aberrational displacement of the direction of a star is due to the velocity of the spaceship. For velocities of 10 to 20 km/sec it is small, for velocities of about 100 km/sec it may reach approximately 1'. If direction finding of the star is carried out, for example, in order to measure the angle between the star and the center of some planet, the aberrational displacement produces an error in this measurement. As a result of this, the vertex angle of the position surface cone will differ from the correct value. Aberrational errors also appear in finding the direction to planets and other celestial bodies.

When measuring the angle between a star and the center of a planet, the proper motion of stars, its motion with respect to other stars, gives rise to a displacement of the axis of the conical position surface. The problem is that the ship's computer has in storage stellar coordinates corresponding to some particular time, which will not coincide with the moment of direction measurement. The larger the difference in the times and the higher the proper velocity of the star, the larger the difference will be between the actual star coordinates and those used in the astrotracker. Consequently, the axial displacement of the conical position surface from its true position will also be larger.

It is true that the proper motions of the stars are small. For example, the annual motion of Sirius is 6.6 μrad**; Arcturus, 11.4; Procyon, 1.6; Altair, 3.3; Pollux, 3.1; Fomalhaut, 1.8; Regulus, 1.22. Other stars have, as a rule, a lower proper velocity, but even these low velocities may lead to quite large errors in the determination of the position surface. For example, a spaceship is 10 million km from the earth, and to determine a position surface, the angle is measured between the center of the earth and the star Arcturus. Assume the ship's computer has in it stellar coordinates which were calculated for a time which differs from the measurement date by half a year. The error in the linear displacement S of the conical position surface will be:

$$S = \Delta\alpha D,$$

where $\Delta\alpha$ is the variation of the position of the star during half a year in radians, and D is the distance to the earth in km.

* Abberation of light is the astronomical term for the deviation of the apparent position of a star on the celestial sphere from its actual position. This deviation is due to the finite propagation velocity of light and to the motion of the observer. For more details on this, see, for example /7/.

** A microradian is equal to 10^{-6} radians. (One radian is 57° 17' 44.8".)

Substituting the values of $\Delta\alpha$ and D, we obtain

$$S=\frac{11.4}{2}\cdot\frac{10,000,000}{1,000,000}=57\text{ km}.$$

This error by "cosmic scales" is of course, small, but the distance of the spaceship from the earth in the example was relatively small. For a distance of 100 million km from the earth, the error will be 570 km, and if, in addition, the stellar coordinates are wrong by 1 year, the error increases to 1140 km. Such an error apparently will be noticeable.

Both this and the previous error, caused by aberrational displacement, are systematic, and therefore can theoretically be compensated for with sufficient equipment.

An error in the determination of the position surface may also arise in finding the direction of binary stars due to the unequal brightness of the two stars. In this case, in measuring the angle between a planet and a star, there may be errors in both the position of the cone's axis in space and the vertex angle of the conical surface.

In a number of cases, the distance between the components of binary stars are quite large. Thus, for example, the components of the binary star in Ursa Major are situated about 12' apart.

Binary stars are not so numerous. Statistics show that of stars up to the 9th stellar magnitude, about 5.5% are binary, but for the brighter stars the percentage of binaries increases. At the present time, very detailed catalogues exist. The simplest method for combatting this error is to choose ordinary, nonbinary stars for generating position surfaces.

Another possible reason for errors in the determination of the spaceship's position surface is the parallactic star displacement. [Parallax is the apparent displacement of an object as seen from different points.] If, for example, the angular coordinates of stars on the celestial sphere are calculated under the assumption that the observer is at the center of the sun, another observer far from the sun will not see the stars at "their usual places," but they will appear slightly shifted. Parallactic displacement results in a change in the orientation of the cone's axis and in the angle of the cone when the angle between a star and the center of a planet are measured.

Measurement errors in determining the angle between the centers of planets may result from noncoincidence of the center of brightness with the geometrical center of the planet, inaccurate knowledge of the distance between the planets and the finite propagation velocity of light, as well as errors in the instruments used.

The displacement of the brightness center depends on many factors, but mainly on the phase of the planet. This may cause an error in the determination of the direction of the planet's center of several minutes of arc and even tens of minutes in the case of short distances to the planet.

The absolute distances between celestial bodies are known to within 0.01%. If the distance between the planets is 20 million km, the possible error in the determination of the distance between them is 2000 km. This creates an error in the determination of the position surface.

Because of the finite propagation velocity of light, in the time it takes light to reach the spaceship from the planet, the planet moves along its orbit and its coordinates vary considerably. For example it takes a light ray 480 seconds to travel 143 million km from Mars. In this time, Mars moves along its orbit more than 11,000 km.

In order to eliminate this error, it is necessary to give the computer the coordinates of the planet corresponding to the time of the observation. It must also be given the time it takes a light ray to traverse the distance from the planet to the ship. However, it is not so simple to carry this out in practice. The distance to the planet is unknown, since the exact position of the ship at this moment is unknown. As a matter of fact, the measurement is being made just to determine this position. In the first approximation, apparently, it is necessary to use the position of the ship given by the previously calculated orbit of its motion.

Measurement of the angular diameter of a celestial body is possible in several ways. For example, it is possible to use a movable circular screen (Figure 81) whose plane is perpendicular to the direction of the center of the celestial body.

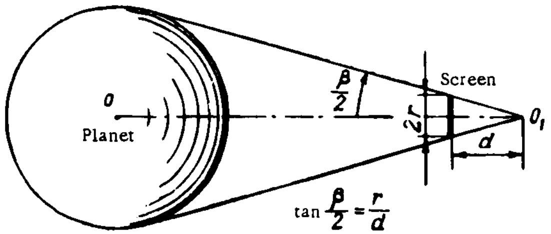

FIGURE 81. Measurement of the angular diameter β of a planet by means of a movable circular screen of radius r

A screen of diameter $2r$ moves perpendicularly to the axis OO_1 to a position where the edges of the celestial body are barely visible. The measurement process can be easily automated by means of photocells and a servomechanism which controls the distance d and the direction of the plane of the screen.

As can be seen from the figure, the angular diameter of the celestial body is:

$$\beta = 2 \arctan \frac{r}{d}.$$

Measurement of the apparent angular diameter of the celestial body is also possible by tracking the edge of its disk (Figure 82) by means of one or several telescopes.

As was shown above, measurement of the apparent angular diameter of the celestial body makes it possible to determine a spherical position surface of the spaceship. The error of the determination of the position surface by this method depends on the measurement error of the apparent angular diameter β and on the accuracy of the known diameter.

The diameters of the planets are known to within 0.01%. Errors in these measurements are due to the influence of the planets' atmospheres, their nonsphericity, nonuniform illumination of their surfaces, as well as instrument errors in the measurement system.

Measurement of distances to celestial bodies is possible by radiation and radar methods. The radiation method is based on the relationship between the amount of radiant energy falling on a heat-sensitive element and the distance to the energy source.

According to calculations of V. P. Seleznev /29/, the relative error of measurement of the distance to the sun by this method is 0.01 to 0.1%. The absolute error of the distance measurement is approximately equal to the diameter of the earth in flights near Venus, Mars, or Jupiter and reaches 5 million km at a distance from the sun corresponding to the apogee of Pluto (7 billion km).

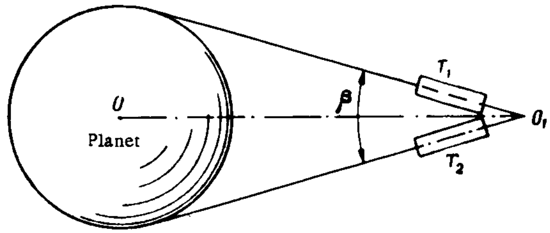

FIGURE 82. Measurement of the angular diameter β of a celestial body by following the edge of its disk by means of the telescopes T_1 and T_2

A variation of the intensity of the radiation flux of a heavenly body may lead to large errors in the measurement of its distance. This is a fundamental disadvantage of this method.

The radar method of distance measurement consists of sending a radio signal to the celestial body and measuring the time interval between the sending of the signal and the arrival of the reflected signal. This method is widely used now in aviation. Radio-altimeters, mounted on all modern airplanes, are used for measuring the distance to the surface of the earth by radar.

An important disadvantage of this method is the need for a large source of electric energy on the ship, since a very powerful signal has to be sent to remote celestial bodies. It can therefore be assumed that radar will be used only in the neighborhood of celestial bodies.

The errors in the measurement of the parameters and in the calculation of the coordinates of celestial bodies cause the errors in the determination of the position surfaces of a spaceship.

The error in the determination of the ship's position depends on the errors of the determination of the position surfaces and on the angle of their intersection. The closer this angle is to a right angle, the smaller the error. Consequently, it is necessary to choose methods which make it possible to determine the position surface with a higher accuracy, and, in addition, methods and celestial bodies which provide the most favorable intersection angles of the position surfaces.

Suppose at a given moment the ship is situated at the point M (Figure 83). In order to determine its position, its azimuth α, height (elevation angle) h, and distance D are measured from the point K. All these parameters are measured with some errors: $\alpha_{meas} = \alpha + \Delta\alpha$; $h_{meas} = h + \Delta h$; $D_{meas} = D + \Delta D$

(where α, h and D are the actual parameters of the spaceship, at the given moment, and $\Delta\alpha$, Δh, and ΔD are the respective measurement errors).

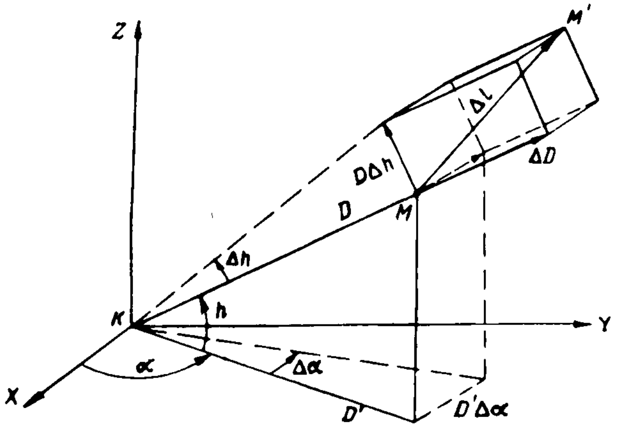

FIGURE 83. Error of the position determination of a spaceship from the azimuth α, height (elevation angle) h, and distance D, all measured from the point M

In this case the calculated position of the ship is at the point M'. The error Δl of the determination of its position is equal to the distance between the points M and M':

$$\Delta l = \sqrt{(\Delta D)^2 + (D\Delta h)^2 + (D\Delta\alpha)^2} = \\ = D\sqrt{\left(\frac{\Delta D}{D}\right)^2 + (\Delta h)^2 + (\Delta\alpha)^2}.$$

As can be seen from the formula, the error in determining the position of a spaceship depends both on the measurement errors and on the distance to the ship; the larger the distance, the larger the error. For this reason, the angle-distance method cannot be used successfully to determine the position of an interplanetary ship at a large distance from the earth.

Even at comparatively small distances to the spaceship (1 million km) and small (1') angular measurement errors of α and h, the error in determining the position of a spaceship due to only these two primary errors will be about 420 km.

Consequently, during the main phases of the trajectory "on board" navigation methods, methods in which the ship's position is determined from measurements of some parameters directly from the spaceship, will be used.

In the above example, all the three position surfaces, plane, cone, and sphere, intersect at right angles, and the error of the position determination of the ship is minimal. However, the position surfaces may not intersect at right angles as shown in Figure 83. Then, even if the measurement accuracy of the parameters determining the position surfaces is high, a large error is possible in the determination of the ship's position. In such cases other parameters should be used to provide more favorable intersection angles of the position surfaces. One of the important problems of space navigation is choosing those methods of locating the orbit of the spaceship which, on a given section of the trajectory, provide the highest accuracy.

Two successive positions of a spaceship make it possible to determine its velocity vector. Suppose a ship is successively situated at the point M_1

at time T_1 and at the point M_2 at time T_2 (Figure 84). The absolute value of the velocity vector is then

$$V=\frac{S}{T_2-T_1}=\frac{\sqrt{x_2^2+y_2^2+z_2^2}}{T_2-T_1}$$

and its direction is

$$\tan\gamma=\frac{y_2}{x_2};$$

$$\cos\beta=\frac{z_2}{S}=\frac{z_2}{\sqrt{x_2^2+y_2^2+z_2^2}}$$

In this way, the mean value of the velocity on a rectilinear section of the orbit is determined. The method is therefore suitable only when the spaceship is far from a gravitational center, which corresponds to a low-curvature orbit with a nearly constant velocity. Note that for the same errors in determining the ship's position, the errors in the calculation of the velocity vector decrease with longer time intervals between the position measurements.

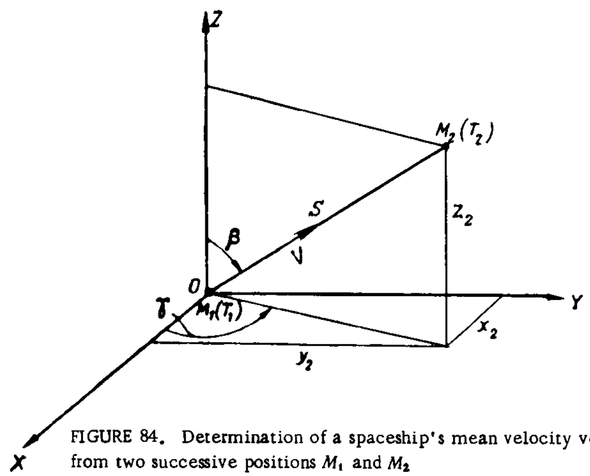

FIGURE 84. Determination of a spaceship's mean velocity vector V from two successive positions M_1 and M_2

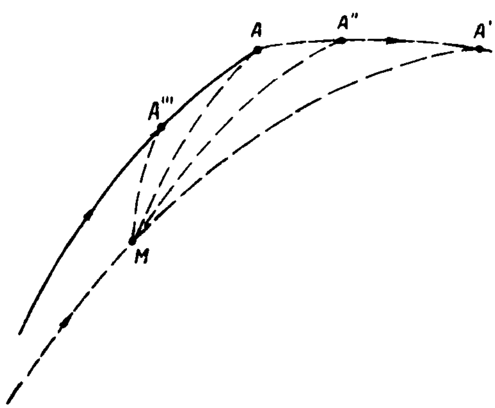

FIGURE 85. Possible solutions of the problem of correcting a spaceship's trajectory to arrive at the correct orbit, starting from the point M of the actual trajectory

Let us try to estimate the accuracy that the distance S between the two positions should be so that the calculated velocity would have a given accuracy.

Suppose it is necessary to determine the velocity of a spaceship with a relative error not larger than 1%. Let the distance between the successive positions of the ship be 10,000 km. We want to find the required measurement accuracy of the distance ΔS. Since $\Delta V/V = \Delta S/S$, we have $\Delta S = S(\Delta V/V) = 10{,}000 \cdot 0.01 = 100$ km.

Consequently, the error in the velocity determination will be not larger than 1% if the distance is measured with an error up to 100 km.

A velocity determination accuracy of 1% is not high compared with the required accuracy of the position determination of the interplanetary ship. To measure a distance of 10,000 km with an error not larger than 100 km is quite a difficult problem, and therefore this method of determination the velocity of a spaceship is not highly accurate. We shall give below other possible methods for more accurate measurement of the velocity of a spaceship.

The known positions of a spaceship, its coordinates at certain moments, make it possible to calculate the elements of its orbit. For example, the current rectangular position coordinates of a spaceship are functions of the six elements of its orbit $(i, \Omega, \omega, a, e, T)$ and of the time t:

$$x = f_1(t, i, \Omega, \omega, a, e, T);$$
$$y = f_2(t, i, \Omega, \omega, a, e, T);$$
$$z = f_3(t, i, \Omega; \omega, a, e, T).$$

It is obvious that the stated problem, calculation of the oribtal elements of a spaceship, can be solved if the coordinates of the ship are determined by some method for two times: x_1, y_1, z_1 for time t_1, and x_2, y_2, z_2 for time t_2. Having a system of six equations with the known quantities $x_1, y_1, z_1, t_1, x_2, y_2, z_2, t_2$ and the six unknown orbital elements it is possible to determine the orbital elements.

The velocity components of a spaceship along the axes of a chosen coordinate system are also functions of its coordinates, the elements of the orbit, and the time. Therefore, by determining the coordinates of a spaceship at some moment and its velocity components $\left(V_x = \frac{dx}{dt},\ V_y = \frac{dy}{dt}\right.$, and $\left.V_z = \frac{dz}{dt}\right)$, it is possible to calculate all the elements of its orbit. Other methods for determining the orbital elements of a spaceship are also possible.

An important problem in interplanetary navigation is the correction of the actual orbit if the deviation from the preassigned orbit exceeds the permissible value in the given conditions. For this purpose we will use the coordinates of the spaceship obtained by solving the position problem.

In principle the following methods of correction of the actual orbit are possible. Suppose at point M (Figure 85) the crew of a spaceship detected a deviation from the prescribed trajectory. If the actual orbit will later intersect the desired one, it may be decided to follow the former orbit without any maneuvers until meeting the calculated orbit at the point A'. They may also decide to carry out maneuvers in order to arrive at one of the points of the calculated orbit on the section AA', for example A''. It is also possible to go to the nearest point of the planned trajectory (the point A) where transition to another orbit was envisaged. Finally, it is possible to go directly to the calculated orbit (the point A''').

The choice of the method of arriving at the correct trajectory depends on the particular conditions and the magnitude of the deviation. But regardless of the method, any transfer orbit will be either elliptical, parabolic, or hyperbolic with respect to the given celestial body. From the viewpoint of minimum fuel expenditure, it is expedient to transfer to the desired orbit along elliptical paths.

A point to be remembered in solving the problem of orbit correction is that at the point where the ship reaches the desired trajectory (points A', A'', A, A'''), an appropriate maneuver must be carried out in order to continue the motion along the correct orbit.

Finally, we point to another important aspect. The importance of timing in space flight and the necessity of keeping to the flight plan along the predetermined trajectory have already been mentioned. Consequently, it is necessary not only to transfer to the correct trajectory, but also to reach it at the scheduled time. Consequently, in moving along transfer orbits to the desired trajectory, the spaceship should follow a rigorously determined schedule.

Between two points of space in the gravitational field of a celestial body there exist, in general, an infinite number of elliptical orbits. However, for a given velocity only one of them can be traversed in a given time. Thus, the complete solution of the correction problem consists in finding an orbit between the position of the spaceship at a given moment and the chosen point of transfer to the correct trajectory along which the ship will reach the planned point on the correct orbit at the correct time.

These are the basic principles of the solution of some problems of space navigation, analysis and correction of motion trajectories.

§4. Navigation Systems and Spaceship Equipment

Let us now consider some of the most representative present and possible future systems and equipment for determining the position of spaceships.

For determining the position of artificial earth satellites, the Americans use a phase-measuring system called "Minitrack." Its operation is based on measuring the phase difference of high-frequency signals from the satellite's transmitter and received by antennas situated some distance apart.

The equipment consists of the satellite's transmitter, a set of ground receiving stations, and a "Vanguard" computational center with an IBM-704 computer, interconnected by a communication system.

Let us consider briefly the principles of its operation. At the points A_1 and A_2 (Figure 86) the antennas are mounted. The distance between them corresponds to n wavelengths emitted by the satellite transmitter, or a $360°n$ phase angle. The distance to the satellite is considerably larger than the distance between the antennas, and therefore it can be assumed that the distances from the satellite to the points A_1 and P are equal. At these points the phases of the signal emitted by the satellite transmitter are identical, while A_1 and A_2 differ. The segment PA_2 is the difference between the distances traversed by the signal from the satellite to the antennas. Suppose it contains m wavelengths, or a $360°m$ phase angle of the signal. Then the direction of the satellite, determined by the angle α, can be found from the relation

$$\cos\alpha = \frac{PA_2}{A_1A_2} = \frac{m}{n}.$$

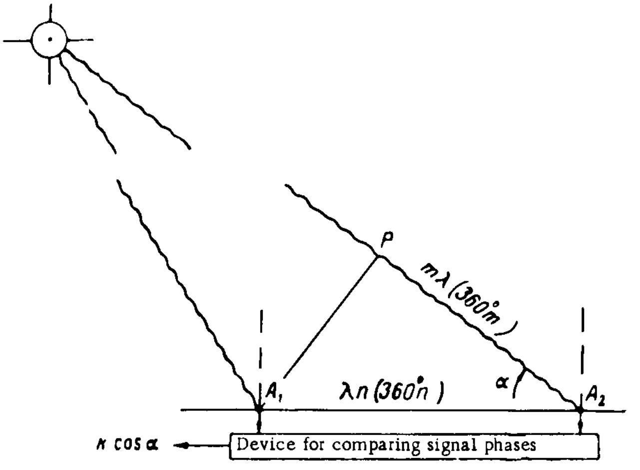

FIGURE 86. Operational principle of the radio-electronic system based on the measurement of the phase difference of signals from a spacecraft

α - the direction of the satellite.

Thus, by measuring the phase difference of the signal from the satellite at the points A_1 and A_2 the direction α of the satellite can easily be determined. However, for each measured phase difference m, there also corresponds a series of other possible positions of the satellite. Rigorously speaking, as we already mentioned, the measured phase difference gives the position surface of the spacecraft in the form of a hyperboloid of revolution of two sheets, whose axis coincides with its base line. The position surface is the same in this case. The assumption that the distance to the satellite is considerably larger than the length of the base line makes it possible to replace this complicated position surface by a simpler one, a two-sheeted circular cone with an axis coinciding with the base line. The generators of the cone coincide with the asymptotes of the hyperboloid, and the opening of the cone is equal to 2α.

One position surface does not make it possible to determine either the coordinates of the object, or even its direction. The "Minitrack" system uses a second set of antennas to determine the direction of the satellite. The base line of the first antenna system lies on a north-south line, and the base line of the second system on an east-west line (Figure 87).

Structurally, the antenna systems are multielement grids whose reception patterns cover an azimuth angle of up to 90° and a height (elevation angle) of up to 12°. Altogether the receiving station has eight grid antennas. Four of them, with base lines of 150 m, are arranged cross-wise in the north-south and east-west directions. The other four form three pairs, two of which serve to eliminate the ambiguity in the north-west direction, while the third eliminates the ambiguity in the east-west direction.

The radio signal has a phase repetition every 360°, and an ambiguity in the angle α therefore results if the difference in the distances from the satellite to the points A_1 and A_2 exceeds one wavelength. This uncertainty is eliminated by an additional arrangement of antennas on each base line, so that the shortest distance between two antennas does not give a phase difference larger than 360°.

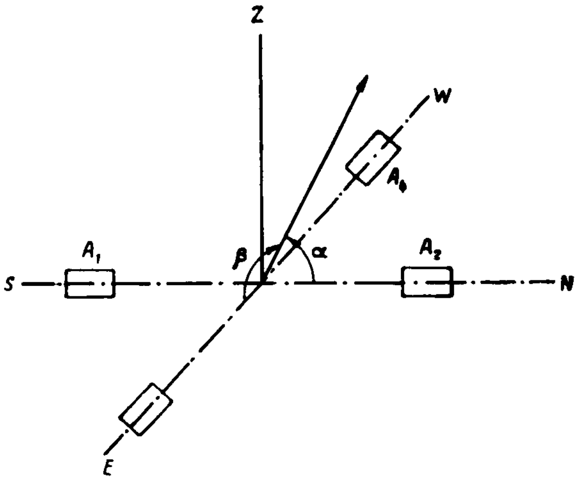

FIGURE 87. Positions of antennas A_1, A_2, A_3 and A_4 of the "Minitrack" system; α and β are angles determining the direction of the satellite

Thus, the ambiguity in the measurement of the phase difference is resolved and the system measures two parameters, the angles α and β with respect to the base lines. To each parameter corresponds a position surface in the form of a two-sheeted circular cone with an opening of 2α and 2β. The axes of the conical position surfaces coincide with the base lines, and therefore also intersect at an angle of 90°.

As a result of the intersection of the four cones, eight possible satellite directions are obtained. Which of these is the true direction to the satellite? It is obvious that four directions are at once excluded, since they lie below the horizon. The reception pattern of the antenna systems, a rough estimate of the position of the satellite, observation of its motion during some time, and a number of other factors, make it possible to exclude the remaining false directions and accurately determine the direction to the satellite.

The "Minitrack" system permits a determination not of the position of the satellite, nor of its spatial coordinates, but only of its direction at a given moment. It is clear that the position of the satellite can be determined by means of two such systems, simultaneously observing the satellite from different points on earth.

The satellite's velocity components, which are determined from the dependence of its angular position on the time, aid in calculating the elements of the orbit. The components are obtained by tracking the satellite during some period.

To track satellites with orbit inclinations of 30 to 40°, a network of nine "Minitrack" stations was created. These are located approximately in the meridional direction in the USA and South America. The stations are connected to the computational center by a communication system. The precalculated positions, the satellite ephemeris, obtained as a result of the processing of the observations by a computer, are transmitted to various observation points.

A new system, based on the "Minitrack" has been developed. The "Azusa" system directly determines a spacecraft's position in space. It includes a range finder based on the principle: inquiry (from the earth)—answer (from the spacecraft). This third parameter, the range, gives a position surface in the form of a sphere. The intersection point of the direction, determined by the angles α and β, with the sphere gives the position of the spacecraft.

The range of the spacecraft is determined by superimposing on the signal of the range finder's transmitter a series of modulating frequencies, which are produced by the vehicle's transponder. The phase of the signal received on the earth is compared with the phase of the transmitted signal, and determines the range. Several modulating frequencies are used in order to raise the accuracy of the range measurement. The lowest frequency is used to determine distances in a wide range, and the higher frequencies provide data within this interval.

Initially it was intended to use the "Azusa" system for guiding the "Atlas" ICBM. It is reported in the foreign press that with small modifications, the equipment of the system will make it possible to determine the position of rockets intended for lunar flight.

For tracking ballistic rockets and satellites one American firm proposed a "Sortey" electro-optical system (Figure 88). A photomultiplier tube is placed in the focal plane of the lens of a camera with a narrow slit in front of it. By means of a rotating mirror (3), the image of a section of the sky is scanned, alternately in the planes XOZ and YOZ. By means of a synchronizing device, coupled to the rotating mirror and the output of the photomultiplier, the increment of the angular displacement of the rotating mirror is determined in order to obtain the exact angular distance between two light sources, the spacecraft and some star. The exact angular position of the star, its azimuth and elevation, is determined by means of an IBM-7090 computer. Apparently such a system must provide a tracking accuracy of the order of 1" for an angular rotation velocity of the mirror of 20 rpm.

The "Sortey" system, as considered by foreign specialists, permits the accurate determination of the azimuth and elevation angles of a spacecraft. The direction to the vehicle is the line of intersection of two position surfaces, a plane, corresponding to the azimuth, and a circular cone, corresponding to the vehicle's elevation angle. The use of two systems, situated at different points on the terrestrial surface, makes it possible, by simultaneous measurement, to determine correctly the position of a vehicle in space.

As an example of a navigational system on board a spaceship, we consider the block-diagram of an astrotracker. This is an automatic device for determining the position of an interplanetary ship by means of three celestial bodies, the sun and two planets. The instrument solves the problem in a heliocentric coordinate system.*

* The tracker diagram is taken from /29/.

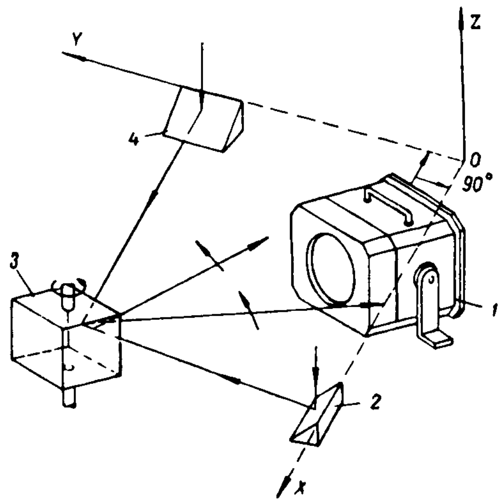

FIGURE 88. Diagram of an electron-optical system for tracking spacecraft

1- case of the main optical system; 2- stationary mirror for scanning along the X axis; 3- rotating mirror; 4- stationary mirror for scanning along the Y axis.

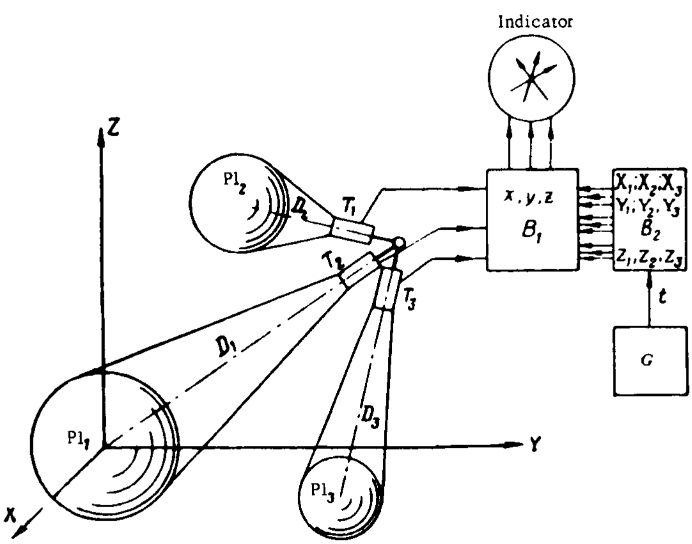

FIGURE 89. Block-diagram of an astrotracker for a spaceship, based on the measurement of some parameters of three celestial bodies

B_1 and B_2- computers; G- stable-frequency oscillator.

The astrotracker contains photoelectric tracking systems, T_1, T_2, and T_3 (Figure 89), which measure either the angular diameters of the celestial bodies or the intensity of their radiation. The corresponding signals from the photoelectric tracking systems proceed to computer B_1 for determining the coordinates of the spaceship at a given time. The same computer is fed with the time-varying coordinates of the centers of the celestial bodies. Computer B_2 calculates the current coordinates of the centers of the celestial bodies: the time is measured by the stable-frequency oscillator G. The current heliocentric coordinates of the ship are obtained as a result of the solution of the problem. They can be displayed by the indicator and introduced into the flight control system in order to compare the actual with the calculated orbits, and help correct the flight trajectory.

The three parameters measured by the astrotracker, the angular diameters of the celestial bodies or their radiation intensities, result in three spherical position surfaces. These generally intersect at two points. Consequently, the solution of the problem by the astrotracker will give the coordinates of two points, one of them, the position of the spaceship at the given moment. Due to the large distances between the points, the incorrect one is easily eliminated by comparison with the calculated trajectory.

On the "Vostok" earth satellite, a navigational globe was used. With it, the pilot could at any time determine his position with respect to the surface of the earth. The globe was mounted in the pilot's cabin in the central part of the instrument panel (Figure 90).

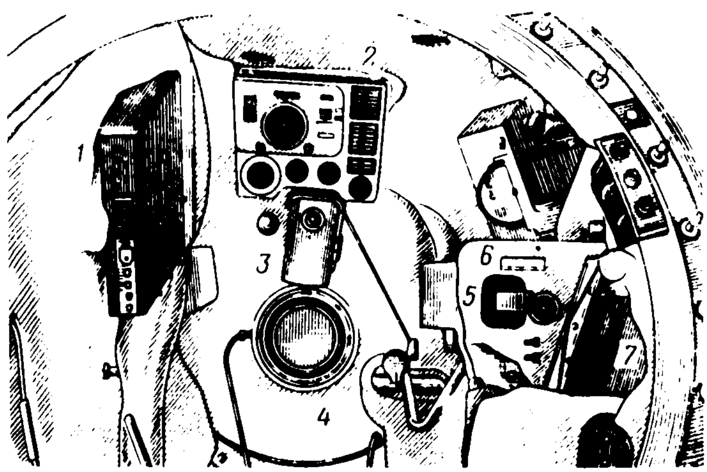

FIGURE 90. Internal view of the cabin of the pilot of a "Vostok" earth satellite

1- pilot's panel; 2- instrument panel with the navigational globe; 3- television camera; 4- window with an optical driftmeter; 5- orientation control handle of the ship; 6- radio reciever; 7- food container.

The design of such an instrument for satellite navigation can be based on simulation of the satellite's motion with respect to the earth, including the daily rotation about its axis. Let us examine a possible scheme for the solution of this problem (Figure 91).

A geographical globe G is rotated by means of motors M_1 and M_2 with respect to two axes, the axis of the earth's daily rotation NS and an axis CC, perpendicular to the orbital plane of the satellite. With respect to the NS axis the globe rotates from west to east with an angular velocity equal to one revolution per stellar day (23 hrs 56 min 04 sec of mean solar time). The motion of the satellite along the orbit with respect to the surface of the earth is shown by an arrow. To replace this motion by a rotation of the globe with respect to the CC axis, the rotation should take place in a direction opposite to the orbital motion of the satellite. The angular velocity of this rotation should apparently be given by the relation:

$$\omega_{cc} = \frac{360^\circ}{P},$$

where P is the period of revolution of a satellite around the earth.

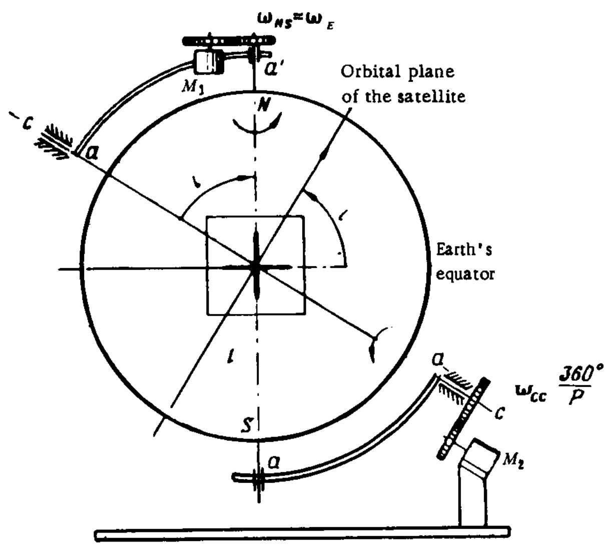

FIGURE 91. Possible kinematic diagram of a navigational globe for determining the position of a satellite with respect to the surface of the earth

G- globe M_1 and M_2 - motors; NS- axis of the earth's rotation; CC- axis perpendicular to the orbital plane of the ship; aa' - arcs for setting the orbit inclination.

The rotation axis CC of the globe is perpendicular to the orbital plane of the satellite, and therefore, as can be seen from the figure, the angle between the NS and CC axes should be equal to the inclination i of the orbit.

We now place in front of the globe a glass scale with a reticule in the form of crossed lines over the center of the globe. Thus, the cross will indicate the position of the satellite with respect to the surface of the earth at any time.

Before use, the globe should be placed in its initial position, the cross should be placed over the entrance point of the satellite into the orbit, and the device should be switched on at the moment it passes this point. When the satellite is launched into an orbit with another inclination, it is necessary to change the length of the arc aa' accordingly.

A constant angular rotation velocity of the globe about the CC axis corresponds, rigorously speaking, only to a circular orbit of a satellite. When moving an elliptical orbit, the angular velocity of the satellite with respect to the center of the earth is not constant, hence, errors appear in the determination of the position of the spaceship. These errors are periodic, and their maximum value is proportional to the eccentricity of the orbit. Orbits of manned earth satellites lie in a comparatively narrow range of heights, between 140 to 160 km and 300 to 500 km, and therefore the eccentricities of their orbits are small.* The resulting errors in the ship's position calculated by this instrument due to the constant rotational velocity of the globe with respect to the CC axis will be small.

In principle it is possible to allow for this error. For this, the angular rotational velocity of motor M_2 should, at each moment, equal the angular rotational velocity of the satellite with respect to the center of the earth. This problem can be solved by means of a programmed device which controls the operation of motor M_2.

A second possible error in determining the position of a satellite is nonallowance for the precession of its orbit; but this error too can, in principle, be compensated for. We previously said that the orbit precession is manifested in a slow rotation of the plane of the orbit with respect to the axis of the earth in a direction opposite to its daily rotation. Consequently, to take into account the orbit precession, the globe should rotate with respect to the NS axis with a velocity equal to the sum of the angular rotational velocity of the earth and the mean precessional angular velocity. Allowance for the precession is particularly necessary for multi-revolution satellites, since the error in the determination of the ship due to nonallowance for the precession accumulates during revolution.

The globe is a completely independent instrument, its operation does not depend on any devices on the ground or aboard ship. Therefore, it can also be used in ground command points for finding the position of the satellite with respect to the surface of the earth at any moment. The instrument can also be used in flight preparation to study the ship's trajectory with respect to the earth, to choose the orbital inclination and entrance point into the orbit in accordance with the purposes of the flight and, finally, to plot the ship's trajectory on a map. For these purposes, however, it is more expedient not to adhere to the real time scale and rotate the globe with respect to both axes with a higher angular velocity.

We note in conclusion the possibility of correcting the readings of the navigational globe from data of other, more accurate systems of determining the position of the satellite. The correction should be carried out by rotating the globe so that the point under the cross will be that point of the terrestrial surface which corresponds to the ship's position according to the data of the more accurate systems.

* The perigee height of the orbit of "Vostok I" was 181 km, the apogee heightswas 327 km, which corresponds to an orbit eccentricity of approximately 0.01.

§5. Inertial and Astroinertial Navigation Systems for Spaceships

An on-board inertial navigation system can be used to determine the position of a ship in space and, what is particularly important, its velocity. In general lines, the operation of inertial systems is based on measuring the accelerations which the spaceship experiences and subsequently, integrating them with respect to time. As is known, the first integration of the measured acceleration gives the velocity of the spaceship, and the second integration gives the path traversed from some initial point, which determines the position of the ship in space. The accelerations are measured by means of accelerometers oriented in a known direction.

The main advantages of inertial systems are their complete independence from off-ship equipment, and the ability to determine directly the position and velocity of the spaceship.

In this section we consider the fundamentals in creating inertial navigation systems of spaceships and the future growth of such systems. Readers wishing to study this problem in more detail are referred to the book "Navigational Devices" by V. P. Seleznev /29/.

One of the basic elements of inertial systems is the accelerometer. The sensitive element of any accelerometer is some inertial mass m connected to the accelerometer case. Its operation is based on measuring the displacements of this inertial mass with respect to the case, or the force acting on it. These parameters are proportional to the acceleration of the spaceship.

Accelerometers can be divided into linear and pendulous accelerometers, depending on the connection between the inertial mass and the case. In the linear accelerometers, the inertial mass moves along a straight line, the sensitive axis of the accelerometer (Figure 92a). The forces, which act on the inertial system due to accelerations of the spaceship along the sensitive axis, are measured by a spring. In pendulous accelerometers (Figure 92b) the accelerations are measured by the angle of deflection of the pendulum from its unperturbed position, in which it is held by a spring.

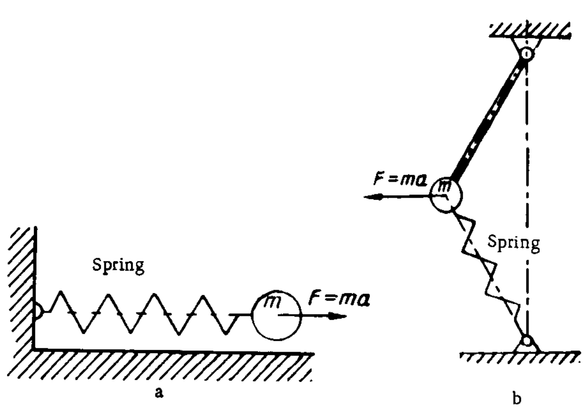

FIGURE 92. Fundamental diagram of a linear (a) and pendulous (b) accelerometer

Let us examine the operation of the simplest linear accelerometer. One end of the accelerometer spring is attached to the body of the ship, the other end to the sensitive element (Figure 92a). Suppose that no forces act on the ship and it moves uniformly and rectilinearly in space (Figure 93a). The sensitive element of the accelerometer moves together with the spaceship with the same velocity V_0. There are no forces stretching the spring, so the signal from the output of the accelerometer is zero.

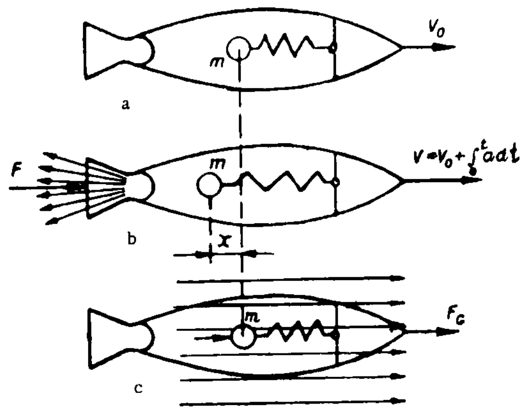

FIGURE 93. Diagram of the operation of an accelerometer on a spaceship

a- in the case of uniform motion; b- in the case of accelerated motion due to the thrust F; c- in the case of accelerated motion due to gravitational force F_G.

Let us now assume that some force F, for example, the thrust of the rocket engine, is applied to the ship (Figure 93b). The ship then acquires an acceleration a_s:

$$a_s = \frac{F}{m_s},$$

where m_s is the mass of the ship.

The acceleration causes an increase in the ship's velocity. However, the sensitive element of the accelerometer tends to maintain its uniform motion with the velocity V_0 and therefore the distance between it and the point of attachment of the spring to the body of the ship increases. As a result a deformation of the spring appears:

$$x = \frac{m}{k} a_s,$$

where k is the stiffness of the spring.

As can be seen from the formula, the deformation of the spring, being the output signal of the accelerometer, is proportional to the acceleration of the spaceship.

Let us now assume that the ship moves only under a gravitational force (Figure 93c). In this case a force $F_G = m_s g$ acts on the spaceship, and a force $F'_G = mg'$ acts on the sensitive element of the accelerometer. The accelerations of the ship and of the sensitive element will be different, since their masses are different; but this difference is so small that it can be disregarded. Consequently, both the ship and the sensitive element will have the same acceleration, that of a freely-falling body. For this reason there is no displacement of the sensitive element of the accelerometer with respect to the ship, the accelerometer's spring is not stretched, and the accelerometer reads zero.

Thus, in spite of the accelerated motion of a spaceship in a gravitational field, an accelerometer based on the measurement of inertial forces does not detect this acceleration. This creates some difficulties in the development of inertial systems for spaceships, but by no means excludes their use for space navigation.

To overcome this difficulty, a signal proportional to the acceleration due to gravitational forces is produced by a special computer. It is fed, together with the output signal of the accelerometer, into the integrators of the inertial system. Thus, the accelerometer records the accelerations which are due to the thrust of the spaceship's engines and to the drag of the atmosphere, and the computer calculates the accelerations due to gravitational forces. Such systems are sometimes called gravity-compensated inertial navigation systems.

In gravity-compensated inertial navigation systems, the velocity components and the distances traversed are computed along the three axes of some coordinate system fixed in inertial space. These may be ecliptic, heliocentric, equatorial, ship-centered, and so on. The sensitive axes of the accelerometers should coincide with the directions of the axes of the chosen coordinate system.

The chosen accelerometer orientation can be maintained either gyroscopically, astronomically, or by inertia-pendulum stabilizers. A coordinate system connected to the spaceship does not require the use of an attitude maintenance system.

It is possible to compensate for gravitational accelerations by placing the accelerometers in a plane perpendicular to the gravitational force (in a horizontal plane). Such inertial systems are called locally level. They are mainly intended for terrestrial aircraft and ships.

The basic elements of inertial navigation systems are:

accelerometers for measuring the accelerations of the spaceship;

a stabilizing system which maintains the accelerometers in a definite orientation with respect to the chosen coordinate system;

a computing device for integrating the accelerations (for determining the velocity, the current coordinates of the ship, the compensating signals, and, in a number of cases, some other parameters);

displays of the output parameters (velocity, coordinates);

a data input for the initial quantities and data (initial coordinates of the ship in the chosen coordinate system);

control devices, power supplies, and some structural elements.

Let us consider some of the possible types of inertial systems for solving the navigational problems of spaceships.

Inertial navigation system with autocompensation for gravitational acceleration. The principle of autocompensation consists of using the results of the determination of the spaceship's coordinates, obtained by the inertial system itself, for calculating the signals which compensate for gravitational acceleration /29/. The system operates as follows.

A gyroscopic stabilizing system (Figure 94) maintains the platform with the accelerometers A_x, A_y, A_z in the XOY plane of a rectangular equatorial coordinate system whose origin is at the center of the earth. The sensitive axes of the accelerometers coincide with the directions of the axes of the coordinate system chosen. The signals from the output of the accelerometers together with the respective compensating signals proceed to the input of the first integrators I_{1x}, I_{1y}, I_{1z}. Integration with respect to time yields the velocity components of the spaceship along the axes of the chosen coordinate system. For a zero initial velocity of the spaceship:

$$V_x = \int_0^t a_x \, dt;$$

$$V_y = \int_0^t a_y \, dt;$$

$$V_z = \int_0^t a_z \, dt.$$

Integrating the obtained velocity components with respect to time in the second integrators (I_{2x}, I_{2y}, I_{2z}) gives the current coordinates of the spaceship:

$$x = x_0 + \int_0^t V_x \, dt;$$

$$y = y_0 + \int_0^t V_y \, dt;$$

$$z = z_0 + \int_0^t V_z \, dt,$$

where x_0, y_0, z_0 are the geocentric coordinates of the launch point.

The current coordinates of the spaceship are introduced into the computer for calculating the signals which compensate for the gravitational acceleration. The velocity components are sent to another channel of the computer for calculating the spaceship's current velocity:

$$V = \sqrt{V_x^2 + V_y^2 + V_z^2}.$$

It is also possible to determine the direction of the velocity vector. (The corresponding formulas are left to the reader.) The data characterizing the current values of the velocity and of the coordinates of the spaceship are indicated by the displays and are sent to the automatic flight control system for subsequent solution of the trajectory correction problem.

Let us examine the compensating signals in the case of flight in the space near the earth. In this case, it is necessary to take into account only the gravitational field of the earth. The perturbing accelerations due to other celestial bodies can be neglected.

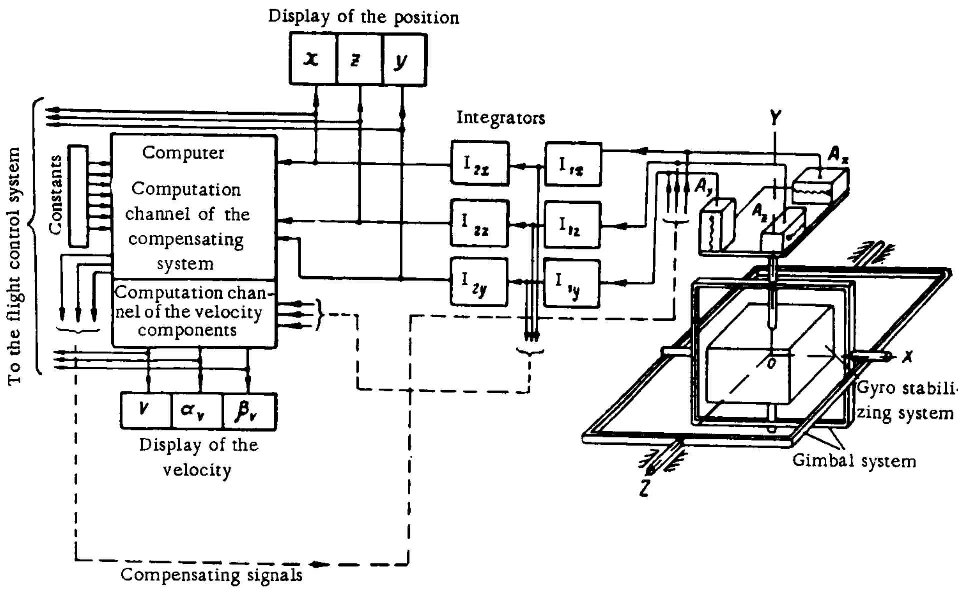

FIGURE 94. Operation of an inertial system with autocompensation for gravitational acceleration

The acceleration components of a spaceship due to the attraction force of the earth F are:

$$g_{x, y, z} = \frac{F_{x, y, z}}{m},$$

where $F_{x, y, z}$ are the respective components of the force of attraction on the ship by the earth's gravitational field, and m is the mass of the ship. But since

$$F_{x, y, z} = \frac{F}{\cos \alpha_{x, y, z}};$$

$$\cos \alpha_x = \frac{x}{r}; \quad \cos \alpha_y = \frac{y}{r}; \quad \cos \alpha_z = \frac{z}{r},$$

where r is the distance from the center of the earth to the ship, we have

$$F_x = \frac{Fx}{r}; \qquad F_y = \frac{Fy}{r}; \qquad F_z = \frac{Fz}{r}.$$

Substituting $F = f\frac{Mm}{r^2}$ where M is the mass of the earth, we obtain

$$g_x = \frac{fMx}{r^3}; \qquad g_y = \frac{fMy}{r^3}; \qquad g_z = \frac{fMz}{r^3}.$$

Thus, the compensating signals should be proportional to the magnitudes of the g_x, g_y, g_z, and the gravitational constant and the mass of the earth must be in the memory of the computer. The values of the current coordinates

x, y, z of the ship are continuously and automatically updated, and the value of r is calculated by the computer. Obviously,

$$r=\sqrt{x^2+y^2+z^2}$$

For interplanetary ships, it is more convenient to use a rectangular heliocentric coordinate system with the XY plane coinciding with the ecliptic. This coordinate system was described in the first chapter. In this case, it is necessary to take into account the gravitational force not of a single celestial body, as in the previous case, but of several celestial bodies (the sun and planets), and the structure of the compensating signals will be as follows:

$$g_x=f\sum_{i=1}^{n} M_i \frac{x-X_i}{r_i^3};$$

$$g_y=f\sum_{i=1}^{n} M_i \frac{y-Y_i}{r_i^3};$$

$$g_z=f\sum_{i=1}^{n} M_i \frac{z-Z_i}{r_i^3},$$

where M_i are the masses of the celestial bodies; X_i, Y_i, Z_i are the heliocentric coordinates of the celestial bodies (for planets they vary in time due to the motion with repsect to the sun); r_i is the distance from the i th celestial body to the spaceship: $r_i=\sqrt{(x-X_i)^2+(y-Y_i)^2+(z-Z_i)^2}$.

The system's errors, in general, are determined by the inaccuracy of the initial coordinates, the calculation error of the compensating signals, drifting of the gyroscopes, and instrument errors of all the units of the system. Investigations show that errors in the formation of the compensating signals are the reason for the system's instability. In other words, the errors in the determination of the coordinates do not damp out, but increase during the operation of the system /29/. This can be seen from the graphs of the variation of the ratio of the current error Δy to the initial error Δy_0 as a function of the altitude above the earth and the flight time, calculated for a ship moving in the direction of the Y axis (Figure 95). In this case the system is stable with respect to the remaining two channels X and Z. However, when moving along an arbitrary trajectory, the system is unstable in all three channels. The decrease in the growth rate of the error with increasing height is due to the decrease in the effect of the gravitational force with increasing flight height.

The instability of such a system is its fundamental shortcoming, but it cannot be concluded that inertial systems cannot be used for space navigation. First, the errors grow relatively slowly and second, the system can be made stable by giving it some external navigational information. Inertial systems can be used not only for space navigation purposes, but also for simulating a spacecraft's trajectory in the gravitational field of any number of celestial bodies. In fact, knowing the orbital parameters of these celestial bodies and given the initial conditions (point of entrance into orbit, and the spacecraft's velocity vector at this point), it is possible to obtain from the computer of such a system the time-variation of the coordinates of the vehicle, i.e., a simulation of its trajectory. In this case, it is also possible to obtain the trajectory taking into account the thrust of the vehicle's engines. For this purpose, at the appropriate moments, analytically calculated data corresponding to the accelerations due to the thrust of the engines should be introduced into the computer.

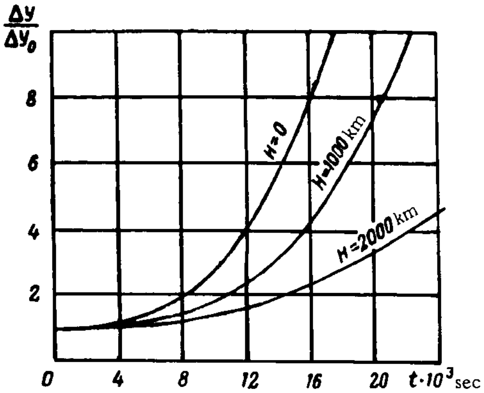

FIGURE 95. Relative error of the y coordinate of a spaceship versus flight time for various altitudes in the case of a ship moving in the direction of the Y axis

Thus, such a system can be used as a simulator for the solution of particular n-body problems and for the selection of optimum flight trajectories of interplanetary ships and spacecraft.*

Inertial navigation system without a stabilization system. We said before that a necessary element of every inertial navigation system is a stabilization system for orienting the accelerometers with respect to the axes of the coordinate system, or, in other words, for representing on the spaceship the axes of the chosen coordinate system. However, it is possible to build an inertial system without such stabilization.

The direction of the axes of the chosen coordinate system can be calculated analytically from the output of six accelerometers, rigidly mounted on the body of the ship. There are two orientations of the sensitive axes of the accelerometers with respect to the coordinate system, which is fixed to the ship. In the first type (Figure 96a) the sensitive axes of the accelerometers are in pairs, perpendicular to the respective coordinate axes. In the second type (Figure 96b) they coincide with the respective axes.

If, due to some external or internal forces, the ship begins rotating, for example, about the Z axis, with some angular acceleration, the sensitive elements of the accelerometers A_{y_1} and A_{y_2} (Figure 96a) shift in opposite directions parallel to the X axis. Signals of the same sign appear at the output of the accelerometers, since the springs of both accelerometers are extended or compressed. On the outputs of these accelerometers a signal appears when the ship accelerates along the X axis, but the signs of the signals will be, as can be easily seen, opposite. Consequently, by comparing the signs of the output signals of the corresponding

* This proposal was for the first time made by V. A. Bodner and V. P. Seleznev /8/.

accelerometers, it is possible to distinguish angular from linear acceleration.

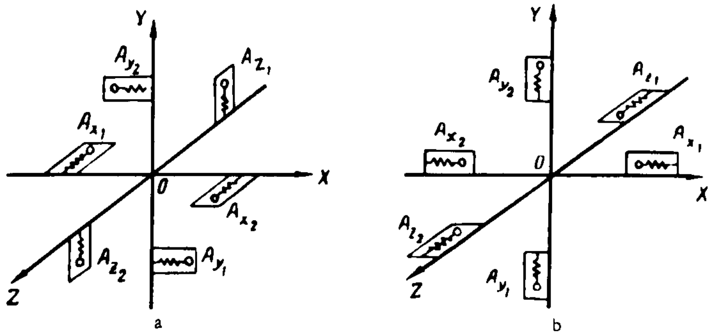

FIGURE 96. Arrangement of a six-accelerometer strapdown inertial system

a- perpendicularly; b- parallel to the coordinate axes.

Accelerometers placed along the coordinate axes (Figure 96b) measure the angular velocities of the spaceship. Linear acceleration, for example along the Y axis, causes signals of different signs of accelerometers A_{y_1} and A_{y_2}, and an angular velocity about the Z axis causes signals with the same sign at these accelerometers. Thus, comparing the signs of the accelerometers' outputs makes it possible to determine whether the outputs result from a rotation of the spaceship about its center of mass, or from linear acceleration.

Double integration with respect to time of angular accelerations, as well as integration of angular velocities, determines the angular coordinates or the rotation angles of the coordinate system of the spaceship with respect to the fixed coordinate system, in which the navigational problems are to be solved.

The linear acceleration measured by the accelerometers can be projected on the X_0, Y_0, Z_0, axes of an inertial coordinate system, and then integrated as in the previous system to obtain the ship's velocity components and position in the chosen inertial coordinate system. A direct integration of the signals of the accelerometers with subsequent projection of the results on the axes of the inertial coordinate system is also possible.

As in the stabilized navigator, the obtained coordinates are used for calculating the compensating signals for gravitational accelerations. These signals in the first case are introduced directly into the integrator, and in the second case, are first projected on the axes of the ship's coordinate system.

Errors in the determination of the velocity and position of a spaceship by a strapdown inertial navigation system may be due to errors of the initial orientation of the ship's coordinate system with respect to the fixed inertial system, measurement errors of the accelerometers, errors in calculating the compensating signals, and errors of the integrators. The larger the distance between the accelerometers, the higher the sensitivity of the system to angular motions, but in this case errors may appear in the measurement of the accelerations, including violation of the parallelism of the sensitive axes as a result of deformations of the ship's body. Even small deformations under large distances between the accelerometers may lead to very large errors in the measurement of the accelerations.

The strapdown inertial navigation system with autocompensation for gravitational acceleration is also unstable, i.e., the errors of the coordinate calculation increase with time. It can be used as a "memory" device, in conjunction with the rest of the ship's navigational equipment, to determine its position.

Inertial navigation systems on spaceships make it possible to measure the velocity very accurately, but considerable errors accumulate with time in the determination of the position. Astronomical systems, based on measuring parameters of heavenly bodies, can determine the position of a spaceship very accurately, but the velocity, calculated from the distance between two positions of the ship, results in large errors. The combination of these systems into a single complex navigation system, called an augmented inertial system, as is shown by investigations /29/, not only makes it possible to compensate for the individual disadvantages, but also increases the accuracy of the navigation.

As an example of an augmented system, we consider the astroinertial system, an inertial system to which external information, in the form of elevation angles of celestial bodies, is sent.

Astroinertial system. The system consists of a gyroscopic platform with three accelerometers whose sensitive axes are along the axes of the chosen coordinate system, an astrotracker for tracking three celestial bodies, integrators for calculating the velocity and coordinates, and three computers (Figure 97).

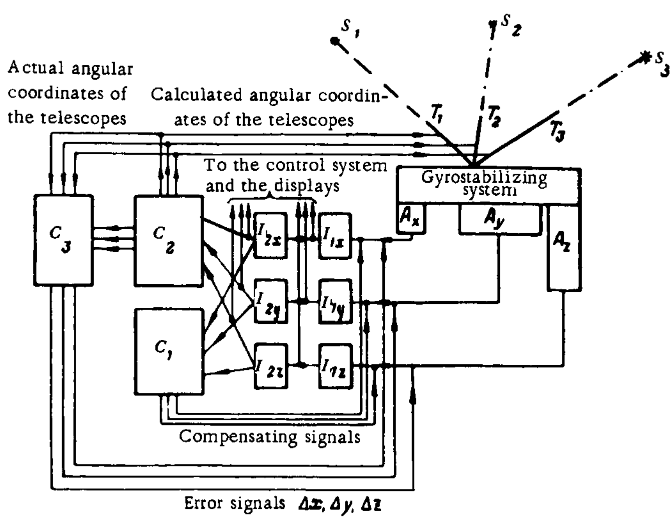

FIGURE 97. An astroinertial navigation system

C_1, C_2, and C_3- computers; T_1, T_2, and T_3 - telescopes.

The coordinates of the spaceship go from the integrators to computer C_1 for calculating the gravitational compensation, and to computer C_2 for obtaining the angular coordinates in order to aim telescopes T_1, T_2, and T_3.

Due to inevitable errors in the calculation of the ship's coordinates, the calculated and actual angular coordinates will not be equal. The difference of the angular coordinates is measured by the telescopes, and fed to the computer C_3, which calculates corrections Δx, Δy, Δz to the ship's coordinates, obtained by integration of the measured accelerations. These corrections are fed back to the acceleration integrators.

Investigations /29/ show that by proper choice of the transfer functions of the loops, it is possible to render the system's position error not only stable with any oscillation frequency of the input errors, but also attenuating. That is, the errors in the initial value of the coordinates of the spaceship will decrease with time. However, on the whole, the system's errors in determining the ship's coordinates cannot be smaller than the errors introduced by the astrotracker, and therefore, in building astroinertial systems, one should strive at building as accurate an astrotracker as possible.

The astroinertial navigation system can work in three modes: the normal operation mode, in which the oscillation period of the determination errors of the ship's coordinates is close to the revolution period of the ship about the celestial body with respect to which the flight is made; a mode of forced elimination of the determination errors of the coordinates at a given time, and, finally, a "memory" mode, when for some reason direction finding of the celestial bodies is impossible, and the system operates as a pure inertial navigator.

The combination of an inertial system with correctors, sources of external information, makes it possible to build a navigation system which can determine with high accuracy both the velocity and position of an interplanetary ship.

Integrated navigation systems for spaceships. Recently, reports have appeared in the foreign press /38/ about the development of a new type of complex for airplane navigation. It includes various navigational information units: an inertial system, airborne radar, radio-electronic means for determining the position of the airplane, astrotrackers, and systems and means for measuring the flight parameters, altitude, velocity, course, and so on.

Such systems differ from the ordinary airplane navigation systems. For example, units, which in given conditions provide the highest accuracy for some navigation problem, are connected automatically. All the aggregate parts of the system are checked periodically and defective equipment is automatically disconnected and reserve equipment is turned on. The outputs of the units are processed together and sent to the displays. The basis of such systems is the integrated control unit, and hence such systems are called integrated.

Excellent reports of such systems are given in the foreign press. Thus, the American AN/ASN-24 integrated airplane navigation system with a computer using semiconductor instruments (weight of the computer with the integrated control unit 14.5 kg) can solve a series of navigational problems, including the determination of the airplane's current geographical coordinates with an error of only 0.07% of the distance traversed /29/.

Some scientists believe it is possible to create integrated systems for the solution of the principal and by-problems of the navigation and flight control of spaceships. In a prolonged spaceflight, failure of individual elements of the navigational equipment is quite likely. This demands periodic checking. The requirement of high flight accuracy along a prescribed trajectory in a prolonged flight and a number of other important problems indicates the experience of using such systems on spaceships.

§6. Some By-problems of Space Navigation

By-problems of space navigation is the term we will apply to auxiliary problems, solved along with the basic problems of navigation considered above. Of these problems the most important are the maintenance or stabilization of the ship in a definite prescribed orientation in space, and the measurement of time in a spaceflight.

The solution of the stabilization problem is primarily necessary for target-directed maneuvers, connected with the modification of the flight trajectory such as descent, transfer to a new orbit, modification of some elements of the flight orbit, and so on. To be able to perform these, as we already know, the spaceship has to be given a definite position in space with respect to some coordinate system.

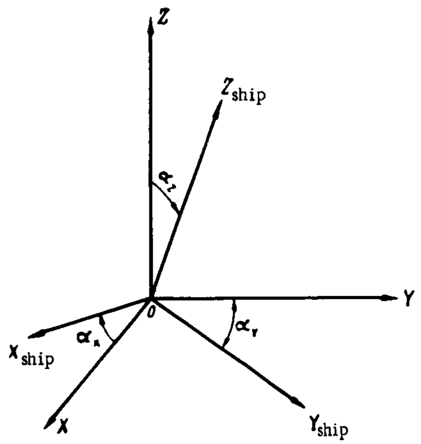

FIGURE 98. Orientation of a ship in space with respect to some XYZ coordinate system

Stabilization of a spaceship is also necessary for other reasons. For example, navigational measurements, communication with the earth, scientific observations, use of solar energy for recharging of electroenergy sources on board, and a series of other circumstances also require the stabilization of the ship in space.

The stabilization of a spaceship is performed with respect to some coordinate system, and therefore to solve the stabilization problem it is first necessary to simulate in some way this coordinate system on the ship. A given position of a ship in space is determined by three angular coordinates (Figure 98), and therefore, some three directions, which in the following we will call stabilization axes, should in general be simulated on the ship. With respect to these axes the spaceship should be stabilized.

We note that for the solution of certain problems it is sufficient to stabilize the ship with respect to one axis only (for example, the Z axis of an earth satellite is made to coincide with the local vertical, and the X and Y axes in the horizontal plane are not stabilized). When an earth satellite or an interplanetary ship, returning to the earth, descends, the X axis should be made to coincide with the orbital velocity vector (in an opposite direction).

Without discussing those devices, by means of which a ship can be stabilized in space, we consider possible principles of simulating the stabilization axes.*

One of the stabilization axes for low-orbit spaceships may be the local vertical. On a ship it can be simulated by various methods, for example, by an inertial system, which navigates in a horizontal coordinate system. In this case the stabilized platform is placed in the horizontal plane, i.e., perpendicular to the local vertical.

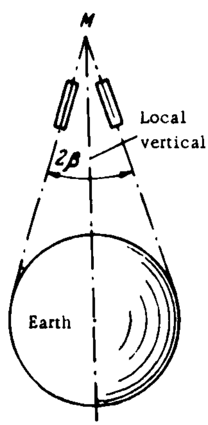

FIGURE 99. Simulation of the local vertical by tracking the edges of the visible disk of the earth (planet)

Simulation of the local vertical is also possible by tracking the visible edges of the earth by means of three or more photoelectric systems, working either in the visible, or infrared part of the spectrum (Figure 99). In this method, for example, a three-beam photoelectric system mounted on the spaceship tracks the visible edges of the earth's disk. The directions of the telescope's optical axes form a trihedral pyramid whose axis coincides with the center of the earth. Thus, the axis of the pyramid is the local vertical at the position of the ship.

Such a scheme presents a series of difficulties; one of these is the difficulty in tracking that part of the earth's disk which is not illuminated by the sun (the night side). Other errors of this method are due to the nonsphericity of the earth, the unevenness of its surface, and the presence of clouds, smoke, etc. When a ship flies above the earth at a height of 500 km, individual sections of the terrain have heights of up to 10 km, and the error in the simulation of the vertical is less than 1°. On the average, the assumed simulation accuracy may amount to a fraction of a degree.

Simulation of the local vertical is also possible by using the screening properties of the terrestrial globe to the isotropic component of cosmic rays.** For this purpose a system of three or four cosmic ray counters, directed to the edge of the visible disk of the earth, are mounted on the ship. If for some reason in the system of four counters, one is pointing

* Problems of the stabilization dynamics of a ship in space and the principles of the stabilization equipment are described in the book of K. B. Alekseev and G. G. Bebenin /3/.

** The isotropic component of cosmic rays consists mainly of neutrons, which are not deflected by the earth's magnetic field.

above the horizon, then due to the screening property of the earth, the number of particles recorded by its opposite counter sharply decreases, and the number of particles recorded by the first counter increases. Tracking of the visible edges of the earth's disk can be automatic.

It is assumed that the accuracy of the vertical simulation by this method is low, about 10°.

The stabilization axis for low-orbit and interplanetary ships returning to the earth may also be the projection of the ship's velocity vector on the horizontal plane. Simulation of this stabilization axis can be most easily done by means of an optical driftmeter, an instrument for observing the relative motion of the earth, due to the motion of the ship. The direction of the stabilization axis coincides with the direction of the motion of the earth. A similar method and corresponding instruments, for example, the AB-52 sights, are used in aircraft navigation for the determination of the vector of the path velocity from the drift angle and its magnitude. On the Soviet "Vostok" spaceships, an optical driftmeter was used (see Figure 90).

In principle, the directions of celestial bodies (sun, moon, planets, and stars) found by appropriate photoelectric tracking devices, can be used as axes of stabilization. Thus, for the descent of a "Vostok" earth satellite, the ship was automatically oriented with respect to the sun /2/.

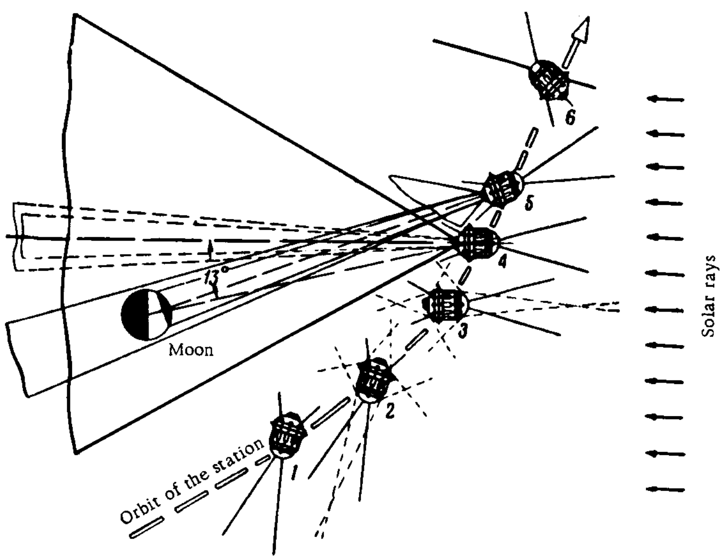

FIGURE 100. Orientation of an automatic interplanetary ship on the sun and moon

1, 2, ..., 6- different positions of the AIS on the orbit (3, 4- orientation of the AIS on the sun); 5- orientation of the AIS on the moon.

The third Soviet lunar rocket, launched on 4 October, 1959, for photographing the invisible part of the lunar surface, was oriented with respect to the sun and the moon (Figure 100). The orientation system turned and stabilized the automatic interplanetary station in the required position first with respect to the sun (positions 3 and 4) and then, for direct photography, in the direction of the moon (position 5).

Errors in this type of stabilization include: inaccurate knowledge of the coordinates of the celestial bodies with respect to which the ship is stabilized, instrument errors in the photoelectric tracking systems, and deviation of the brightness center of the celestial body from its geometrical center. The last reason may lead to considerable errors when orientation is by celestial bodies whose illumination passes through different phases, such as the moon, Venus, and Mercury. The errors will be large when flying near these celestial bodies. For example, when flying near the earth, the orientation on the first or last quarter moon will be in error up to 8', and at closer distances to the moon the error may reach degrees and even tens of degrees.

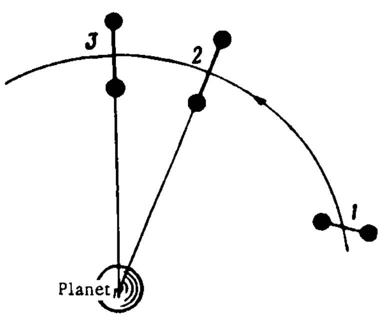

FIGURE 101. Orientation of a dumbbell sensitive along the local vertical

1, 2, 3- positions of the dumbbell on the orbit.

The local vertical when flying inside the sphere of action of any celestial body can be determined by means of a very simple sensitive element which looks like a dumbbell. The rod connecting the massive spherical masses at the ends of the dumbbell will always reach an equilibrium position directed towards the center of the celestial body (Figure 101). This is due to the fact that, when the axis of the dumbbell deviates even slightly from alignment with the vertical to the center of the celestial body, the spherical end closer to the celestial body, experiencing a greater gravitational attraction than the other, will give rise to a moment tending to restore equilibrium.

Incidentally, for this reason the moon always turns the same side to the earth; it is not completely spherical and in this respect resembles a dumbbell.

Such a dumbbell continuously oscillates about the equilibrium position. Reaching the equilibrium position, the dumbbell passes it and continues in the opposite direction, then again returns to it, passing to the other side, and so on. This is how a dumbbell sensitive element will behave in the absence of friction.

When flying along a circular orbit, the oscillation period of a dumbbell sensitive element is:

$$T = \frac{P}{\sqrt{3}},$$

where P is the revolution period of the spaceship around the celestial body.

For an artificial earth satellite on a circular orbit with a revolution period of $P = 90$ min, the oscillation period will be slightly more than 52 min.

It is possible to attenuate the oscillations of the dumbbell. As a damping device it is possible to use, for example, a piston coupled to the dumbbell and moving in a vessel filled with liquid, or a magnetic field. Other simple ways for solving this problem are possible.

Damping of the lunar oscillations is possibly due to internal friction from tides, caused in the body of the moon by the earth.

It should, however, be borne in mind that the forces orienting the dumbbell are small, and therefore it is necessary either to use very massive parts, or a very long rod. The latter is apparently more acceptable, since in a space flight there are no obstacles for placing such a dumbbell outside the ship. Building ships in the form of a dumbbell is also possible. Such a ship will have its Z axis oriented in the direction of the celestial body.

We note, finally, the possibility of simulating the direction of the sun by means of a solar vane. Imagine a light-weight cone of foil or paper with an arm which can rotate about axes perpendicular to the axis of the arm (Figure 102). Such a device, placed outside the spaceship, under the pressure of the sun's light rays will have the axis of the arm pointing in the direction of the sun.

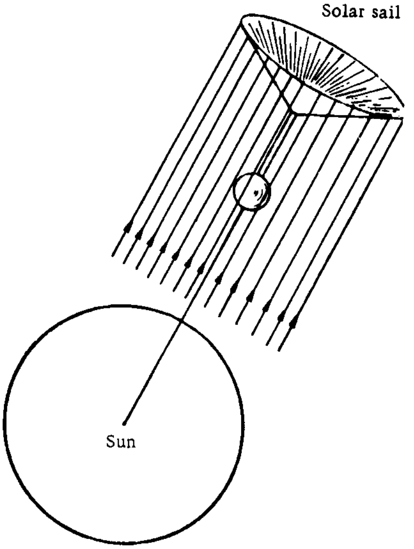

FIGURE 102. Orientation of a sensitive vane element in the direction to the center of the sun

Using a solar vane, it is possible to turn a spaceship. At Saturn's orbit, the pressure of the solar rays is only 0.001 mg/m^2. If the area of the vane is 100 m^2 and the length of the arm is 80 m, a spaceship weighing 1 ton will be turned in the direction of the sun in less than 24 hours.

Another by-problem of space navigation is the problem of autonomous time measurement in space flight. Timekeeping in a space flight can be done by means of various spring clocks, as well as by electric mechanisms. For this purpose, use can also be made of various atomic clocks, high-stability instruments for time measurement. In one of these instruments, a high time measurement accuracy is achieved by a quartz oscillator which is continuously controlled and synchronized with respect to the resonance frequency of the cesium atom (9, 192, 631, 830 cps). The accuracy of the atomic clock is, in relative units, 10^{-9}. Such clocks de deviate in 100 years by only 3 sec. Atomic clocks are heavy: one aircraft model weighs 27 kg /29/. In addition, the failure of such a complicated instrument in a prolonged flight is possible.

In contrast to the atomic clock, spring and electric clocks are completely reliable instruments for time measurement, but for a prolonged flight they are unsuitable due to the considerable errors which increase with time. It is possible to correct these clocks by means of radio signals sent from the earth. However, due to the large distances and to the finite propagation velocity of radio waves, large errors in this method of timekeeping are also probable. The problem therefore arises of working out other methods for keeping and measuring time in interplanetary flights, particularly autonomous methods. One sufficiently simple, and at the same time comparatively accurate method of autonomously measuring time has been proposed by V. P. Seleznev /29/. Like other astronomical methods of time measurement, this method is based on the periodicity of astronomical phenomena.

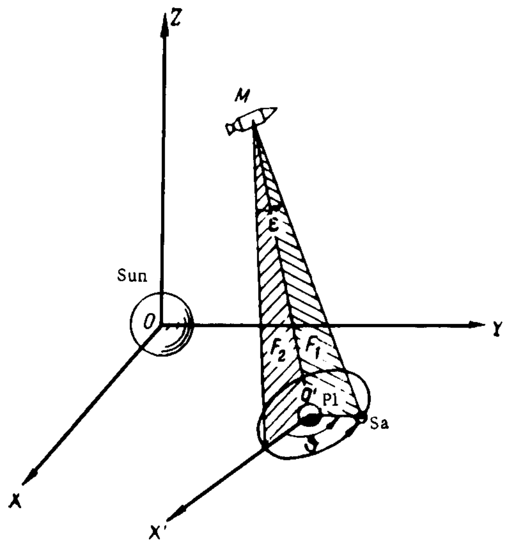

FIGURE 103. Time measurement on a space flight from the position of the satellite Sa of the planet Pl

Let us assume that a satellite Sa of a planet Pl moves along a stationary orbit with known elements (Figure 103). The value of the true anomaly ϑ of the satellite, measured from some initial direction $O'X'$, determines the time interval which passed from the moment it passed the direction $O'X'$. Thus, time measurement by this method requires knowing the moment of passage of the satellite through the line $O'X'$ and measuring the angle ϑ.

The angle ϑ can be measured by a spaceship only indirectly, i. e., by measuring its related dihedral angle ε between the planes F_1 and F_2, by means of which the directions OX' (it may be a direction of a star) and of the satellite Sa are found. The function $\vartheta = f(\varepsilon)$ depends on the mutual positions of the spaceship, planet, satellite, and the orbital plane of the satellite.

The relation between the angle ϑ and the time T can be written in the form:

$$\vartheta = \frac{2\pi}{P}(T - T_0) + f(T - T_0),$$

where P is the period of revolution of the satellite; T_0 is the moment of passage of the satellite through the direction $O'X'$; $f(T - T_0)$ is some correction which is introduced in order to allow for the ellipticity of the satellite's orbit.

In order to perform the indicated measurements and calculate the time, it is necessary to know the coordinates of the centers of the satellite and the planet; their variation with time (the ephemerides of the celestial bodies), the ship's coordinates, obtained from the navigational system, the moment of passage of the satellite through the direction $O'X'$ or the moment of passage through the orbit point nearest to the planet, and the position of the line of apsides of the satellite orbit.

Investigation shows that to increase the accuracy of the time measurement by this method it is necessary to increase the measurement accuracy of the rotation angle of the radius-vector of the satellite, and to choose a satellite (of the sun or of a planet) with as short a period of revolution as possible /29/.

There are many planetary satellites with short revolution periods; both satellites of Mars, six satellites of Jupiter (the revolution period of the sixth satellite is 16.689 days), seven satellites of Saturn (the revolution period of the seventh satellite, Hyperion, is 21.28 days), all six satellites of Uranus, and the first satellite of Neptune, Triton (revolution period 5.88 days).

If an error of 1" is made in the measurement of the angle ϑ and the revolution period of the satellite is 24 hours, then the error in the time determination is only 0.008 sec. For the same measurement accuracy of the angle ϑ, when the moon is used for the time measurement (P = 27.32 days), the error in the time determination is 1.86 sec.

The error in the time determination will be even larger if a planet is used for this purpose. The mean sidereal revolution of the nearest planet to the sun, Mercury, is 87.97 days. For such a revolution period and the same 1" error, in the measurement of the angle ϑ, the error in the time measurement will be more than 6 sec. In view of the fact that an accuracy of 1" in the measurement of the angle ϑ, is very high and for the time being is practically unattainable, the conclusion can be drawn that for the time measurement by this method it is now possible to use only the near satellites of planets.

These, briefly, are the main by-problems of space navigation and the possible methods of their solution.

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